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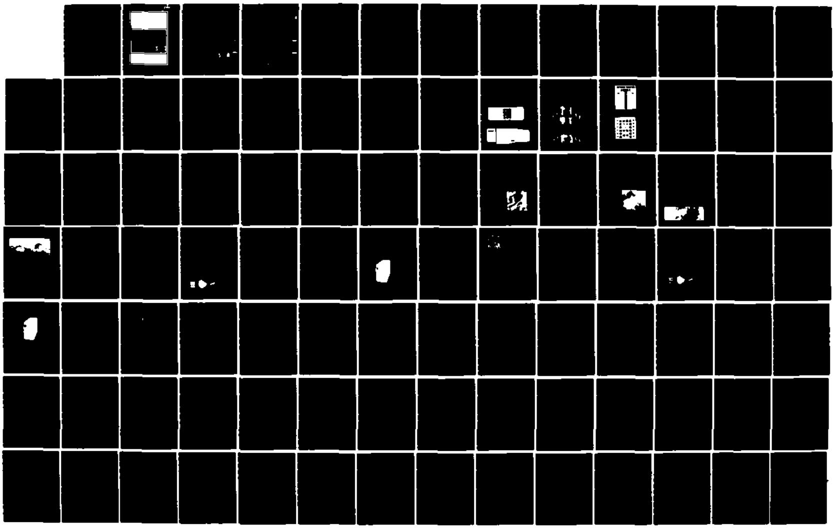

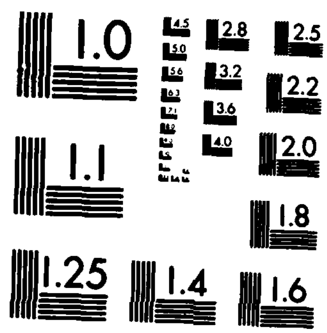
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HELICOPTER GUIDANCE AND CONTROL SYSTEMS FOR BATTLEFIELD SUPPORT

Paper presented at the Guidance and Control Panel 38th Symposium held at the Naval Postgraduate School, Monterey, California, USA, 8–11 May 1984

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PREFACE

The significance of helicopter operations in the land battle scenario of the future is increasing. Advances in sensors, weapons and avionics systems can only lead to improvements in operational effectiveness of helicopters if a total systems approach to the vehicle/crew/mission systems/environment is pursued. These proceedings from

AGARD's ~~The~~ Guidance and Control Panel ~~now~~ address this specific topic since helicopters represent a very significant portion of the future fleet of air vehicles. The various sessions were ~~will be~~ structured as follows: Requirements and operational use of future helicopters; New methods of briefing helicopter crews and tactically setting up the mission systems before and during the flight; Flight control, displays, communications, navigation; Sensors and weapon systems integration in overall cockpit design; and Simulation and flight research and development experience.

The symposium concluded with a Round Table Discussion by the participants.

PREFACE

Les opérations héliportées prennent une importance croissante dans les scénarios de combat terrestres de l'avenir. Pour que les progrès intéressant les systèmes de capteurs, d'armes et d'avionique puissent se traduire par un accroissement de l'efficacité opérationnelle des hélicoptères, il faut aborder les problèmes de véhicules, d'équipages, de systèmes de mission et d'environnement sous l'angle d'une approche global des systèmes.

Les hélicoptères entrent pour une part très importante dans la composition de la future flotte aérienne; c'est pourquoi le Commission Guidage et Pilotage a choisi de traiter maintenant de ce problème spécifique. Les diverses séances seront consacrées aux questions suivantes: Les impératifs des futurs hélicoptères et leur utilisation opérationnelles; Nouvelles méthodes appliquées au briefing des équipages d'hélicoptères et a la mise en place tactique des systèmes de mission avant et pendant le vol; Pilotage, affichage, communications, navigation; Intégration des capteurs et des systèmes d'armes dans l'ensemble du cockpit au stade conceptuel; Expérience en matière de recherche et de développement dans les domaines du vol et de la simulation.

Ce symposium s'est achevé par une Table Ronde à laquelle ont participé tous les délégués.

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The Panel wishes to express its thanks to the US National Delegates for the invitation to hold this meeting in Monterey, CA, USA, and for the facilities and personnel which made the meeting possible.

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*Issued in classified publication CP.359 (Supplement).

*Issued in classified publication CP.359 (Supplement).

TECHNICAL EVALUATION REPORT

by

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1. Introduction

The following summarises the theme of the Conference: "The significance of helicopter operations in the land battle scenario of the future is increasing. Advances in sensors, weapons and avionic systems can only lead to improvements in operational effectiveness of helicopters if a total systems approach to the vehicle/crew/mission systems/environment is pursued".

The helicopter is a flexible element in the land battle, and this flexibility must be exploited by the use of modern guidance and control systems but in an affordable manner. The Keynote Address amplifies this main theme, and the hope of the programme committee was to organise a conference that exposed the current state of the art of weapon and avionic systems for helicopters, and set them in the context of the land battle scenario in order to establish any critical need for research and development into the application of emerging technologies or a need for new technologies. The aim was also to highlight any major challenges or trends in system design.

2. Technical Programme

The programme was organised into five sessions and a final Round Table Discussion. The balance of papers between the nations was particularly good, with a strong technical input from European nations who provided two-thirds of the papers. Only one paper was not presented (Paper 24) and this, it is hoped, will be included in the proceedings. The papers, by country, were:

US	Ca	UK	Fr	Ge	It	Total
8	2	6	5	7	2	30

The meeting was classified NATO SECRET and in the event 5 papers were classified; the rest were unclassified. In particular, Session V was unclassified, and the Round Table Discussion was therefore held in an unclassified environment. This did have the effect of inhibiting the discussion a little.

There was a general concensus that the papers generally were of a high standard and supported the theme of the conference. Most attendees who completed the Questionnaire found the technical programme exceeded their expectations. The theme certainly attracted high interest which supported the need for a conference on Guidance and Control of Battlefield Helicopters.

The one criticism that has been made is that the Round Table Discussion at the end was the only real discussion period of the whole conference. It would have been better if each technical session had time allocated at the end (30-45 minutes) for a discussion on the theme of that session. However, the Round Table Discussion was of a high standard and did draw out the main conclusions and recommendations of the conference. The five gentlemen sitting on the platform: Messrs Fellows (UK), Elliott (US), Roy (UK), Pelegrin (Fr) and Beyer (Ge), all made short but effective points for discussion and the contributions from the floor were apposite and stimulating. The Chairman had no difficulty in maintaining a lively discussion period up to the closing ceremony.

3. Assessment of Technical Sessions

3.1 Session I

Paper 11 by Capt Coyle was a good presentation by an Operator who gave a very "down to earth" presentation of what the operator is looking for in new technology to help in Wartime scenarios. His main message was that a proper dialogue must exist at all stages of a project between procurement specification writers, manufacturers and users. Paper 12 by Messrs Boehm and Haiplik (Ge) was a description of MBB's Air Combat modelling involving guns and missiles. The results clearly depended on the assumptions made, and more debate is needed on these assumptions. Papers 13 and 14 gave a good overview of UK research work in the field and the innovative or even entrepreneural approach to affordable helicopter mission systems which are tailored to the real Army battlefield scenario and operational requirements. The solution to night flying and target acquisition proposed took account of the realistic short range of practical vision in nap of the earth flying, rejecting the costly requirement being suggested by some for 10 km range for infra-red systems.

3.2 Session II

Paper 21 was a useful discussion of the various considerations making up a total mission planning, briefing and debriefing system. The relationship between C^3 network, terrain data base and cockpit display requirements were well described, and ideas for verification of on-line intelligence and data acquisition were given. Paper 22 was particularly good, addressing the data base problems and the presentation/display/ human factor considerations associated with digital maps. The compatibility of colour displays with Night Vision Goggles was also covered.

Paper 23 was a straight description of a commercial development of an electronic map display using flying spot scanner reader of cartographic map with television link to cockpit colour CRT.

3.3 Session III

Paper 31 gave a good summary of the multi-sensor integrated navigation system with combinations of IN, GPS, Omega, Doppler and Magnetic Compass. Paper 32 gave an exposition on the theory and practice of Doppler Navigation: useful for reference. Paper 33 gave a good description of designing down to low cost of a standard AHRS for the three US Services (refits and new projects). The reliability target was ambitious: better than 10,000 hours MTBF.

Paper 34 gave a good description of an integrated mission management system, and made a strong plea for an integrated approach to the system design. Paper 35 gave an interesting description of the design philosophy for the software for high integrity flight control. Various techniques were discussed such as the use of High Order Language (PASCAL) and Assembler Code in parts of the system, and the use of dual programming for high integrity. Paper 36 described a design study and experiments for HF antennae. Data link and voice transmission were studied, and to achieve satisfactory performance at ranges up to 250km, and low reference power (100w), the author concluded that a loop antenna is the ideal radiator for a helicopter. Paper 37 was one of the most significant in the conference. The concept of a fibre optic link between weapon and helicopter is extremely attractive, since it eliminates the need for the helicopter to have a line of sight to the target throughout the flight time.

3.4 Session IV

Paper 41 gave a very good survey of the advantages and disadvantages of roof-mounted, nose-mounted and mast-mounted sights, including all aspects of system design - piloting, vulnerability/detectability, pilot seating position, etc. The conclusion reached was that there was little to choose between roof and nose-mounted sights, but mast mounted sights were good for Air/Ground missions. For Air/Air missions, the choice is open, although there can be disadvantage in the mast-mounted selection. The authors recommended serious consideration of a multi-sensor package unit including Direct View Optics, TV, LLTV, FLIR, Laser Ranger, Reconnaissance Radar, etc. The costs of these elaborate options were not addressed.

Paper 42 discussed various low airspeed measuring devices, including laser anemometry. The requirement was to measure in the range 1 knot to -40 knots and +50 knots in a frequency band 0.05Hz to 2Hz. Flight tests have just begun using a following car with anemometer as the calibration vehicle.

Paper 43 described a 60 G Hz Radar solution to helicopter obstacle warning with a 600m maximum range requirement. Results achieved were greater than 500m in a clutter free environment.

Paper 44 gave a good account of an integrated core avionic system where the author concentrated on Flight Management System, performance and fuel management, plus maintenance parameters. Special emphasis was placed on the need to establish software standards and design procedures.

Paper 45 gave a description of the build-up of a current Data Bus Integrated Avionic System. The navigation sub-system was also on a data bus using a High Order Language and ISA microprocessor. The various sub-systems were connected through microprocessors which were then connected by a high speed data bus. This allows for functional development by the use of the 'intelligent' terminals.

Paper 46 was a very interesting discussion paper starting with the stark history of the number of cockpit controls growing from 40 to 440 in 40 years. The reduction of pilot workload can be achieved by sensor 'fusion' onto electronic display surfaces, eg Map, Threats, Contours and Escape Routes, etc. The way to achieve these advances is through advanced software management, application of artificial intelligence, etc. However, a key problem emerges as to who is responsible for the system design in such integrated systems.

3.5 Session V

Paper 51 gave a good review of the aircrew-aircraft integration problem, and highlighted the design of the sidestick controller. A four axis system is ideal provided the SCAS is reliable. The tests were done on a simulator with a one-on-one air combat task.

Paper 52 described results of a piloted simulator programme to establish control system designs to give good pilot ratings. The analysis of results was carefully done, taking account of limitations in the simulator. The most significant result was that good control system design promoted more aggressive nap of the earth flying - a feature which is very important but hard to quantify.

Paper 53 described a flight simulator for helicopter development. The model of dynamics was very elaborate: the simulator was fixed base with motion onset and buffet with a day/night/dusk visual system. No results of simulation trials were yet available, but the description of facilities was useful.

Paper 54 described a helicopter weapon system simulator in anti-helicopter and anti-tank roles using a variety of weapons such as guns, and for air/air very short range missiles. The pilot and gunner stations were both represented separately, and CGI visual system has been specially developed for these tasks with a terrain data base. As in Paper 53 no experimental results were reported.

Paper 55 provided some insight into why simulator results can be significantly different from flight. Mathematical modelling of the pilot was used to try and explain the discrepancies. The conclusion reached was that the disparity between motion and visual cues was almost certainly the main cause. This paper was a very good simulation applications study, using the NASA Ames Vertical Motion facility with CGI display. The nap of the earth flying tasks were also carefully designed to represent realistic operations.

Paper 56 gave results of flight experiments using a Digital Feel Force System in a BO105 helicopter. The main results given were the desire to have lower breakout spring and damping forces than in existing in-service helicopters.

Paper 57 described theoretical studies of a Gust Alleviation Control System for the rotor blades. The concept involved very complex and difficult engineering realisation, but is a useful guide into possible longer term research and development into improved helicopter ride control.

Paper 58 gave a comprehensive description of flight trials of Night Vision Goggles, Doppler Navigation and Electronic Displays. The results showed that with two crew, the workload was acceptable in night nap of the earth operations. The paper was a thoughtful one with useful discussions on pilot workload assessment techniques.

Paper 59 gave a description of an approach guidance system compatible with ILS using X-band radar and beacon based ground system. The results showed that comparative performance with MLS could be obtained.

3.6 Round Table Discussion

The following points were made by the five men on the platform:

a. The imaginative use of technical solutions already devised for fixed wing aircraft and helicopters should be encouraged. Many of the problems in helicopter operations could be solved if existing technological solutions were properly applied.

b. The target detection and recognition systems in helicopters must be designed as an integral part of the total weapon system. Thus the weapons for helicopters should be optimised specifically to integrate with helicopter systems.

c. Night capability is an essential extension of existing flight and weapon aiming operations: hence the vital importance in developing cost/effective night systems.

d. The helicopter is basically a cheap air vehicle, and so it is vital to do proper cost/performance trade-off studies when designing helicopter mission systems.

e. Crew workload reduction is a high priority objective and this must be achieved in wartime scenarios. However, peacetime training has its own problems and these must also lead to acceptable crew workloads.

f. The way forward into the use of new systems has to be organised in an evolutionary fashion. Large revolutionary changes will lead to high cost of introduction and possible incompatible operational procedures.

g. Maintenance and logistic support must be considered from the outset in developments of new helicopter systems.

h. New and better prediction and assessment techniques are needed to establish true crew workload levels. Are we doing enough work in this area, and have we enough people who are specialists?

i. Present helicopter operations rely too much on speech communications: there is a need for a high integrity data link using self correcting codes, etc.

j. Since computing power is developing rapidly and becoming relatively cheaper, systems using a number of low cost sensors with processing become attractive as an alternative to some of the recent highly sophisticated sensors.

k. The detectability of helicopters is still a major constraint in battlefield operations: more attention to methods of reducing detectability is needed.

l. Simulation is important in developing new systems. Should we be spending more on this kind of activity to help make choices in new system concepts? Helicopter test aircraft are also necessary since some simulation is very difficult and expensive to achieve with sufficient fidelity.

m. In planning battlefield strategy and tactics, the high command must take account of the existence of the helicopters on both sides.

n. One way forward is to do attrition studies with the objective of getting a better balance between helicopters and tanks. There is also need to balance the costs of the two weapon systems, tanks and helicopters.

o. Updating of existing systems is a good way forward rather than just introducing completely new systems (see f. above). Also it is important to get some simple systems into service and then move forward to introduce more innovative ideas.

4. Administrative Arrangements

The meeting site in Monterey was ideal, both for the technical meeting itself and the social mixing of the participants. The US National Authorities responsible are to be congratulated, and in particular Mrs Deanna Zook, the host National Co-ordinator. Any criticisms from participants were very minor indeed. The lecture theatre was very large, but nevertheless the participants did not feel "lost", and the public address system worked very well for most speakers. Because most sessions were classified, no note-taking could be allowed. A number of attendees found this a great disadvantage since no preprints could be distributed. One suggestion was made to group the few classified papers into one session, leaving the majority of sessions unclassified. However, the logical flow of the conference would be lost.

The weather and tourist attraction in Monterey were a great distraction for participants, but most found the informal social arrangements very much better than too many organised events. It was a tribute to the interest of the technical programme that so many attended through the conference, and in particular the final session and Round Table Discussion.

5. Conclusions and Recommendations

Two main areas emerged which represent the key problems that need urgent resolution:

(a) Crew Workload/Man Machine Interface

The very demanding task of nap of the earth flying in European weather conditions and at night poses a major problem. There is still debate as to whether a single crew is at all feasible - even with two crew more research and development is needed to off-load the pilot by the use of automation of 'routine' functions, allowing the tactical flexibility of the pilot to be exploited. More research is needed to measure objectively and quantify work load.

A subsidiary area is the development of electronic displays of sufficient brightness and contrast to operate in a battlefield support helicopter cockpit. Flat panel displays have great advantages in cockpit space relative to CRTs. The input/output controls and cockpit controls need to be improved by the better complementary use of touch displays, multi-function keys, switches and direct voice input.

(b) Cost Effective and Affordable Totally Integrated Systems

There was unanimity in the need to design the total Mission System/Helicopter as a unified exercise, and that this complex task must aim for low cost solutions. This will need radical approaches and ideas compared with some current lines of development. In fact, designing down to some cost target is suggested as a firm discipline.

As well as the two main thrusts needed, the following emerged as very important areas for future research and development:

(c) The helicopter system needs to be designed for a realistic battlefield environment in war conditions. At the same time peacetime training has to be considered (with its additional safety constraints). Also, crews need training and adapting to new system concepts, and to achieve this it is essential that operational front line helicopter crews are involved in the formulative and test stages of research and development of new systems.

(d) To maximise the use of new sensors, blending and integration of sensor outputs and signal processing as well as the development of 'smart' displays is the way forward. New software and VLSI developments will allow the realisation of such developments.

(e) Navigation, flight control and propulsion control will need to be integrated to make better use of redundant sensors to improve system integrity at modest cost.

(f) All these advances depend on the development of fault-tolerant software and better and cheaper methods of software design and validation.

Finally, the advances in technology are so rapid that a conference with similar objectives should be held in about three years time, to review again these critical areas as applied to the battlefield helicopter.

KEYNOTE ADDRESS BY

GEOFFREY C HOWELL, DIRECTOR AIRCRAFT EQUIPMENT AND SYSTEMS DEVELOPMENT
MINISTRY OF DEFENCE (PROCUREMENT EXECUTIVE) UK

I was very pleased indeed to be invited to give the keynote address at this GCP Conference. I was still a member of the panel when this topic of Helicopter Guidance and Control Systems for Battlefield Support was selected, and the importance of this topic has heightened in the intervening two years. I was not surprised to hear from Jeff Fellows, the Chairman of the Programme Committee, that they had had an extremely good response to the Call for Papers, and therefore were able to assemble a good programme for the four days of this Conference. As well as giving this opening talk, I have also agreed to produce the Technical Evaluation Report: this gives me a unique opportunity to speculate as to what I hope will be achieved in this meeting, and then assess what, in fact, was achieved.

The technologies in the Guidance and Control field have advanced very rapidly over the last few years. Most of them are equally relevant to military helicopter systems as for fixed wing combat aircraft. However, trade-offs in cost-effectiveness are considerably different - helicopters in the land battle scenario must have a reasonable cost ceiling compared with other elements such as tanks, armoured vehicles and surface/air missiles. I believe the main problem to resolve in the next five years is to capitalise on existing or certainly emerging technologies in the particular applications to Battlefield support helicopters. All NATO nations are now defining the operational requirements for Light Attack and Utility Helicopters to be in service in the mid-1990s or so, because of the need to exploit the tactical flexibility of the helicopter as a launching platform for weapons as well as the other roles in reconnaissance and transporting men and supplies. In this keynote address I would like to highlight key features that need to be addressed to make helicopter systems contribute in a cost effective way to the land battle.

Helicopter/Weapon Systems Integration

It is well known that most of the current weapon systems fitted to helicopters have been developed initially for ground launching or man portable - guns, rockets, guided weapons. Compromises have had to be made in the sighting arrangements and the physical installation solutions. In the future, the crew/system interface, installation and avionic system integration must be considered from the outset of the design of the vehicle and its guidance and control system. One of the key technologies that needs exploiting is Target Detection and Recognition for line of sight and lock before launch missiles. The sensor package (probably infra-red) has to be designed for surveillance and weapon aiming. A wide angular range is needed and varying fields of view. These requirements must be integrated into a sensible affordable sensor package that is easy to align and ergonomically satisfactory from the viewing and control point of view. The debate still rages between roof-mounted and mast-mounted sights. The complexity and cost of satisfactory mast-mounted solutions needs justifying against any consequent improvement in helicopter concealment. When fire and forget weapons are available the whole launch scenario changes. Similarly, the use of laser designators for missile guidance depends on the weapon system solution. The autonomous system needs a long range laser and very accurate sight pointing accuracy; the co-operative solution on the other hand requires the helicopter with the illuminating laser designator (acting as a substitute for the Forward Air Controller) to be near the target and therefore potentially vulnerable.

Crew Workload/Man Machine Interface

The requirements for future Battlefield Support Helicopters call for a two-man crew to share the piloting and weapon system operator tasks: usually with the proviso that on the injury of one man, the other can fly the helicopter back to base. This calls for a dual-control vehicle. The side-by-side layout is the best for overall crew station design, but obviously leads to a higher drag fuselage than the tandem layout. For the battlefield scenario the pre-flight tactical briefing is vital, and data link/cockpit displays for on-line intelligence updating is a further step in the automation needed in the cockpit. It is my strong belief that the crew station and system integration design is the biggest challenge we have today. Electronic displays, Direct Voice Input, Knowledge-based Intelligent systems, etc, are all essential technologies that need to be harnessed in an intelligent manner to produce the best man/machine interface with the crew wearing full NBC kit for the role of the battlefield support helicopter. A key need in the development phase is to have an adequate dynamic helicopter/mission system simulation facility as well as a flight test vehicle. The development of suitable simulation facilities should be a high priority of Research and Development Organisations in order to be able to mount representative comparative experiments to optimise the man/machine interface, utilising the new technologies well in time to be able to influence the helicopter design.

Survivability and Stealth

The operational missions required of the battlefield support helicopter makes it very vulnerable indeed to all threats associated with the melee of the battle: SAMs, rockets, artillery, small arms fire and attacks from WP helicopters, are simultaneously present. Recent experience has highlighted this problem, and the two distinct lines of defence need pursuing - Improving survivability when hit, and Avoidance of being hit by stealth technologies and tactics. There is a limit on what can be done to minimise the effect of battlefield damage: the cost and weight of modern armour protection is, in itself, limiting. On the systems side, the use of distributed digital computing, multifunction displays and the two-man crew can provide a degree of redundancy which should be exploited more in the future. The hazard of power lines is very real, and development of cable warning systems is another vital area in reduction of vulnerability of helicopter systems in Nap of the Earth operations.

The subject of stealth systems and tactics for helicopters is a growing one and must be of highest priority. Some key technologies are:

a. Nap of the Earth flying techniques requiring high agility from the vehicle/flight control system combination.

b. Reduction of Optical, IR Radar and Acoustic Signatures.

c. Use of ECM techniques both active and passive.

The fitting of self-defence weapons is another key advance that must be pursued. Again these will have to be designed from the outset for helicopter use - particularly for slow speed phases of flight.

Reliability, Availability, Maintainability and Durability (RAM-D)

The logistic support for the helicopter operating in the front line areas is clearly an important issue. A recent UK requirement for a new Army helicopter asks for the maintenance manhours/flying hour not to be greater than one. The advanced battlefield support helicopter of the kind now being specified will have significantly more mission systems than the previous generation, but the availability must be at least as good. Thus the reliability of new systems must be improved considerably, and indeed, some of the more sophisticated and complex systems may have to be rejected on these grounds.

The other strong driver to improve reliability is to reduce in-service costs and hence the overall Life Cycle Costs. There is no doubt that the cost of the helicopter is going to be balanced against other elements of the land battle, such as tanks, SAMs, etc. The unit price must keep within an upper limit of around £500K to £1M - this has to be compared with an advanced combat aircraft UPC of £10M or so (10 to 20 times). This statistic puts into perspective the task ahead of us to achieve effective helicopter weapon systems, using advanced technology, that are affordable.

I have covered rapidly and rather superficially the subject, but I hope I have reinforced the need to approach the Helicopter Guidance and Control Systems for Battlefield Support in a Total Systems fashion. I believe this to be essential in deciding how to make best use of the advances in technology now becoming available. The programme of this Conference looks extremely promising. I am looking forward to all the Sessions, and I hope the final Round the Table Discussion in particular will highlight the key thrust areas for future research and development. I will close with the headlines of areas I believe to be most significant:

Helicopter/Weapon Systems Integration.

Crew Workload/Man Machine Interface.

Survivability and Stealth.

Reliability, Availability, Maintainability and Durability.

Life Cycle Costs: Development, Unit Production Cost, In-service Support Costs.

OPERATIONAL REQUIREMENTS

VERSUS

TECHNOLOGICAL CAPABILITIES

BY
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SUMMARY

This paper discusses the lessons learned in the wartime use of helicopters, and the impact that modern technology can have on what have become common themes. Lessons learned, the constraints on helicopter operations, and the requirements of the battlefield helicopter are integrated to show where technology would better serve the operators. Performance monitoring, handling qualities and radio communications examples are presented, and a brief outline of how existing technology could be used to update an existing helicopter is made.

INTRODUCTION

That the helicopter is poised to undergo as dramatic an advance as the fixed wing aircraft did in changing from piston to jet engines should be obvious to every person in this room, even without the projections of what LHX or JVX will look like or the fanciful cinematics of "Blue Thunder" or "Airwolf". The helicopter is coming of age.

The helicopter's image as a relatively fragile, underpowered device useful for hauling troops, cargo or casualties is gradually being removed. Developments in the technologies of structures, composites, and engines have overcome many previous shortcomings and the helicopter is about to become a device whose uses may well be limited solely by man's imagination.

HISTORY

The whole history of aviation has been one of development based on knowledge. Often this has been hard won and many have paid with their lives for lack of knowledge. Rotary wing aviation unfortunately has not been immune to this costly way of learning, and many of todays advances owe their impetus to the shortcomings of the past. The improvements in airframe design and powerplants and the use of composite materials, for the most part, are logical extensions of the things we have grown familiar with. They present no great leap of knowledge nor difficulty in application. This is not the case in electronics. Even the most expert are having difficulty in keeping up with developments and no one can forsee all the possible uses for many new items. That electronics will do almost anything we ask is not the problem. What do we want done and how much we want to do is of far more importance.

The main improvements that electronics have provided for the helicopter have been in the area of night and bad weather operations. These two areas show up all too readily man's shortcomings, and here electronics has extended capabilities. Wars do not stop at night, nor on account of rain, and up to now this has been a crippling limitation of the helicopter. Yet before we get too deeply into the improvements possible with technology it is necessary to review some of the other, non-technical lessons that have been so dearly learned in the past. While the helicopter has been used in warfare for only a relatively short time in comparison with fixed wing aircraft, many lessons can be learned from WW II, Korea, Malaysia, Algeria, Vietnam, Yom Kippur and the Falklands conflicts.

THE LESSONS OF WAR

Briefly and by no means completely, the history of aviation in general and helicopters in particular in a combat environment seems to point to some interesting aspects:

Operations are conducted beyond what the original design intended in terms of weight, and to a lesser extent, role.

Unit level modifications are the norm, not the exception.

Periods between overhaul are extended

Personnel, both operators and maintainers, are pushed to limits of endurance

Operators and maintainers have low experience levels

To a greater or lesser extent, all of these aspects have been present in the previously mentioned conflicts, and interestingly enough, have parallels in the development in fixed wing aviation. There is no reason to suspect that they will be missing from the next war, and if we can use technological improvements in such a way as to focus on these lessons learned in the past, we may make a significant improvement in our operational capabilities.

RULES, LIMITATIONS, AND LAWS

During peacetime the operation of any aircraft is subject to a never ending number of rules and regulations, and operating limitations. Broadly speaking these items break down in the following way:

RULES AND REGULATIONS

Those regulations set out by authorities to govern the conduct of peacetime operations in a safe and orderly manner.

As every pilot will tell you, there are many rules that you can break with impunity. It is expected that many of the numerous rules and regulations governing day to day operations will be gratefully thrown out when a war comes along. Rules cannot cover the variety of situations encountered in peacetime, and to slavishly follow them in war would be folly of the highest order.

AIRCRAFT LIMITATIONS

The restriction of an aircraft parameter, such as weight, speed, engine power etc., to permit safety of flight, long life and continued use at an acceptable maintenance penalty.

Likewise, limitations are placed on helicopters to preserve them. Speaking from personal experience, it is not difficult to exceed some of these limitations and not know it, or to exceed them with small or no consequence. It only takes one transmission overtorque, with a subsequent discovery that all it means is checking the oil filters, to realize that a healthy margin of safety is built into many components. In my test flying career, I have become more aware of the allowances made for inadvertant excursions beyond the limitations published for the operators.

For example, there is almost no way of determining the effective weight (aircraft plus underslung load) of a helicopter without resorting to cumbersome and imprecise charts. The UH-60, for example, designed to be able to achieve a rate of climb in the hover on a 4000 foot pressure altitude, 95 degree F day, has been flown at 50% above its original design weight. While few would argue with having this sort of capability, consider the consequences - the operational pilot seeing this sort of demonstration now realizes that he can overload his UH-60 with impunity. As previously discussed, there is a very strong possibility that given a war environment he is going to overload the machine anyway, so now caution can be thrown to the winds. Unfortunately there is a third set of constraints, often ignored and much less forgiving, that the operational pilot is about to face, one that he cannot ignore, like rules, or stretch, like limitations.

PHYSICAL LAWS

The laws of aerodynamics, performance, thermodynamics and strength of materials.

There are already numerous examples of what happens when people attemp to break some of these laws - one of the most common being performance required exceeding performance available from the engine. Some services have reacted to accidents of this nature by regulating that prior to every sortie, performance must be calculated using the various charts. As one of the first casualties of a war is the regulations, and as the charts aren't light in weight or easy to use, the operational pilot is put in a very unenviable situation. Suitable technological application in this area would eliminate this problem, however, adequate guidance beyond the limitations prescribed for peacetime

operations must be given. For example, it is feasible to put all the performance charts of a helicopter into a microprocessor, which would certainly eliminate the problem of carrying around all the charts, and probably give more accuracy at the same time. If such a microprocessor were to be hooked up to sensors in the aircraft, such as air temperature, pressure and radar altitude, a low airspeed sensor and engine conditions such as torque, temperature and RPM, it would be possible to automate the performance calculation process. The very large change in power required versus airspeed in the area where current pitot static systems are ineffective is graphically demonstrated in Figure 1. With such an automated system the pilot would be able to instantly determine if he had overloaded his machine (useful in peacetime to satisfy the authorities) and also determine if he has sufficient power available for the more critical wartime situation. Such technology has existed for at least five years, and an example of an attempt to put it into a helicopter is shown in Figure 2. Unfortunately this equipment did not proceed beyond the demonstration phase. There is clearly an operational requirement for this equipment, and the technology has been available, but it has not been harnessed.

It would be relatively easy to programme such a device to cater for the wartime scenario. Given that limitations are going to be exceeded, the computer could say what capabilities exist for the conditions. It could, for example, take into account flight control margins for hovering out of wind, or allow for increased weight when it is possible. Rather than leaving the pilot to "suck-it-and-see" in an overloaded aircraft, some precise guidance could be given. The penalty for failing the "suck-it-and-see" test is very high. Laws, especially physical laws, cannot be broken repeatedly without a very high penalty.

THE OTHER LESSONS

Unit level modifications are going to be the norm, not the exception, as the personnel most affected by shortcomings in material and equipment cannot and will not wait for the official answer to their problems. How this can be improved upon is not an easy question to answer and is beyond the scope of this paper.

The other lessons gleaned from the past should also have an impact on how we use our technology. It has been gratifying to see more and more "on condition" maintenance being demanded and built into helicopter designs, however, do they take account of the very strong possibility that the normal operating limits will be exceeded? Are there data recorded to see exactly what limits were exceeded and for how long? Has anyone thought about how this will impact the long-term life of the aircraft concerned? This will not only have an impact on wartime availability, but also on peacetime operation, when perhaps not all that should be reported is in fact reported.

That everyone is worked to the limit of their endurance in a war is a fact that all who design, build or test military equipment should be well aware of. What it means is that the system that worked so well in the lab or in set field exercises may be too complex or too time consuming to be worthwhile in a war. In addition, it is worth considering the factors of mans adaptability, and his desire to make things work. That man could fly some of the early helicopters and perform some of his magnificent feats of flying in them is surely the only proof that is needed of his adaptability. New equipment often needs a long hard look by the sceptical to see if it really is progress. An inertial navigation system with superb accuracy but with a 10 minute warm-up time is not only a hinderance if you want to take off in a hurry, it could be a liability if it is the only system you have. The increased capabilities that electronics have bequeathed on the helicopter were mentioned previously, and the prime areas affected are night and bad weather operations. Previously these had afforded tired crews at least some chance for a rest. Unless adequate provision is made for around the clock manning for aircrew and groundcrew, the advantages gained will be for naught. Tired people make more mistakes and faster computers will only accelerate the gravity of the mistakes.

MURPHY'S LAW

No endeavour of mans is immune to this law. War may amplify it, due to hard use of equipment and the inevitable fog of war, but even the best engineering will not make it go away. It certainly did not disappear with the advent of the microprocessor. However, wars do not stop because the circuit breaker blows, or because the voltage levels are not right or because the operator inputs wrong information. For example, with a means of distributing information the overall number of errors may decrease slightly, for a given amount of information to be inserted, but the impact of any error is magnified many times. Consider a totally integrated cockpit that has a cassette device for the input of data. Due to the volume of information required, for example, friendly and enemy troop positions, code words, call signs, frequencies, and so on, the compiling of data will probably be done by someone other than the pilot or crew. Without this system, if one person makes an error entering a grid reference in a navigation system the consequences of the error are probably small in the overall picture. If one person makes an error in a data cassette that a whole formation will use, the results could be much worse. At the least, a means of verifying data should be avialable to the crew, and a suitable means of completing the mission after multiple failures should be high on the priority list.

The low experience levels of the operators and maintainers are also cause for concern. At present it takes a relatively short time to train a helicopter pilot, in comparison with his fixed wing counterpart, as most helicopters have been little more than engines, airframes and at best a rudimentary armament system. More complex helicopters require more lengthy training, more specialized pilots and more specialized groundcrew. As we have not yet started genetically engineering pilots it is well to remember that pilots are no better nor worse than they ever have been. It is not enough to cater for the mythical and nonexistant "average" pilot, as by definition, half of the pilots will be below the mark. Having trained and equipped the future aviator at great cost can we afford to put him at risk in the same fashion as the cheaply trained and cheaply equipped?

A REVIEW OF THE CHARACTERISTICS OF HELICOPTER OPERATIONS

At the risk of offending many who are intimately familiar with helicopter operations in the land scenario, a review of the items that separate helicopter operations from those of our fixed wing brethren is necessary. These include, in the main:

OPERATION FROM REMOTE, AUSTERE SITES

Weather information is either limited or non-existent.
Navaids are extremely limited, and air traffic control is almost non-existent (there are also no runways on which to align the compass).
Communications are mostly to and through Army networks.
Knowledge of elevation, variation and so on is limited to what is shown on the maps available.
Operations are conducted in small groups, and the supported units are widely scattered.
Operations are conducted in conditions of poor visibility.

FLEXIBILITY OF OPERATION

The crews must be expert in reconnaissance, observation, artillery fire control, anti-tank operations and numerous other army related specialities, often simultaneously.
Response time to different tasks must be rapid.
Numerous radios must be monitored and different call signs and even different jargon must be used on each different radio.

CONTINUOUS PERFORMANCE MONITORING

Torque limitations.
Engine speed and temperature limitations.
Low airspeed flight envelope.
Low altitude flight with very limited terrain clearance.

There are obviously too many of these to be dealt with completely in this brief discussion, however, some of the implications of these problems should be considered. Let us look at the impact of these aspects on only one new type of system. Personally, one of the pieces of avionics I would like to have in any helicopter is a Doppler navigation system, so for the purposes of illustration let us look at a Doppler navigation system . For operation at low speed or in reduced visibility it gives the ability to observe groundspeed and eases the task of navigating. It is to date the only navaid that I am aware of that will permit an accurate zero groundspeed IFR hover, yet it requires very precise inputs from the magnetic compass to be of any use in navigation. How will the accuracy of the compass be checked when operating behind a barn one day and in a village the next? Even more interesting, how will the magnetic variation be checked for the area of operation? How long will it take to programme the system,and can it be done without starting the engines or APU?

A Doppler navigation system has been used here as it is a personal preference, however, any navigation system is only as good as the person who is using it. It will not show errors in entry, it will not correct mistakes, it will not relieve the operator from checking it, and most important, it will never replace a man with a map. The best that any navigation system can hope to do is to assist in navigation. It will permit the operator to perform another task, such as observe the enemy or fire weapons, instead of constantly referring to a map, but it will not permit him to stop navigating. It certainly will not let the crew be ignorant of the principles and practices of navigation.

There have already been examples of human error defeating very complex, triplicated navigation systems, and these have been without the complications of fatigue, pressure or confusion that will be prevalent in war. Navigation systems are needed, as the job of navigating at night or in bad weather is difficult. Even in these conditions however, it is still necessary to update the system by some other external means. Navigation systems must not be made so excessively complex as to be too difficult to use or understand, nor must they be treated or advertised as replacing

maps. Even moving map displays will still require checking against the ground, which presupposes a minimum skill level at map reading. Any device designed to replace the printed paper map should also be capable of displaying at least as much information and to the same resolution as a map held 18 inches (0.5 metre) away from the eye.

There is an operational requirement to assist in navigation, but there is no requirement to make the operator a slave to the technology.

PERFORMANCE MONITORING AND HANDLING QUALITIES

Moving on to performance monitoring and handling qualities, it is interesting to note the number of engine instruments in fighter cockpits and the number in helicopters. It is also interesting to note that not only are the critical instruments that require monitoring in helicopters more numerous, but they all appear to change simultaneously. (See Figure 3)

The requirement to monitor so many engine parameters simultaneously must be questioned. Each of the parameters concerned can be justified on the basis that it is vital to maintain the longevity of the component, and has historically always been in helicopters, yet why do we not have transmissions that can handle all the power that the engine can put out, and which do not require a separate indication? Why do we not have a single indication that will tell if the engine is performing satisfactorily, and if the power required for the ambient conditions and at other sites enroute is sufficient? Why do we not have more collective lever designs like that of the Gazelle, that incorporate changes in force? Without having to look inside, the pilot is able to pull to a position that is defined by a tactile stop, (that is easily overcome), and to know that no limitations will be exceeded. This is very handy in rapid reaction situations. As air to air combat between helicopters becomes more probable such a feature will be essential.

Helicopters have been saddled with handed down technology for too long, and it is only recently that this mold has been broken. The first generation of helicopters to be fitted with low airspeed sensors has just emerged and it promises to revolutionize the way helicopters will be used and what low airspeed envelopes will mean. The next step is to integrate them into the previously mentioned performance computer which will be able to predict to the pilot what manoeuvers he will and will not be able to perform. For example, current low velocity limitations are most rudimentary and are given as side and rear wind speed limits. Often these have no mention of the effect of weight or density altitude, or if they do mention them require the use of cumbersome graphs. Side and rear wind limitations are based for the most part on having sufficient yaw or pitch control to maintain a desired flight condition. This is not a limitation in the terms of the definition given previously, but it is rapidly approaching being a law.

Like any other aerodynamic variable, the amount of thrust available from a tail rotor or the amount of pitching moment available from a main rotor will vary, at constant RPM and angle of attack, with density altitude. What may be an adequate margin at sea level may not be adequate at greater altitudes. For the most part, no direction is given to the operational pilot on this matter, and the result is another attempt to break a physical law. By using a low airspeed indicator it would be possible to warn the operator when he is approaching an area of inadequate yaw control before he gets himself into trouble. Again, as it is very likely that the helicopter will be operated beyond the peacetime limits, information in this area is also needed. The technology has existed for some time to obtain low airspeed information, but those helicopters that use such equipment do not use it for anything related to performance or control margins.

INTEGRATING THE TALK SHOW

Communication is one other area of great difference between fixed and rotary wing aircraft. It is not uncommon for an Army helicopter to have four or more radios. The agencies that a helicopter may have to talk to are shown in Figure 4. A breakdown of the radio types is given below and should be taken as being a minimum fit.

TYPE	USE
VHF-FM (2)	Talk to Army units
VHF-AM	Civilian air traffic control
UHF-AM	Military air traffic, Forward Air Control and inter-formation
HF	Beyond line-of-sight communication

Not only is there a proliferation of radios, but there are different ways of talking on each one and to further complicate the matter the same aircraft may often

have different call signs on different radio nets. To say that handling the various radios in a battlefield helicopter is one of the more difficult tasks is a large understatement. Until recently this has had to be accomplished with different radio control heads and inadequate intercommunication facilities that required the radio volume to be set at the radio, and that required hands off the flight controls to change transmitters and frequencies.

Recent developments in cockpit design have improved some aspects of this problem, however, they have led to another problem where perhaps we can take some very good lessons from the fixed wing community. Using the AHIP and SH-60 as examples it is interesting to note how all the controls on the cyclic and collective have been positioned for the thumbs to use; none have been positioned for the other digits. This first attempt at Hands On Cyclic and Collective (HOCAC) is long overdue, but there may be room for further improvement. Compare this with the F-18 where the controls have been more equitably divided. Surely more use of the other, equally dexterous, digits of the pilots hands could be found without much more effort. Figures 5 and 6 show these differences.

Returning to communications, careful control of technology is required to truly reduce the workload of the operators. Prior to analyzing what is or is not progress in radios, let us look at what we have at present. Radio packages seem to fit into two catagories, those with preset channels and those without. For an operational pilot faced with a suite of radios, his actions are relatively simple. To change frequencies he either moves the preset channel knob to another channel, or if he has no preset channels he dials up the new frequency. To change the transmitter he simply changes the position of the selector, and to select any receiver he merely changes the position of the appropriate switch. To see what frequency he is transmitting on he need only look at the control head of the radio concerned. This system has its shortcomings, mostly in the panel space it takes up, however, it is simple to operate and exceptionally easy to learn to use. Any technological change should, if at all possible, improve on the situation, and certainly should not either take up more panel space or, more important, require more steps to preform a task. As we shall see this is not always the case.

Two examples of radio controllers and intercom systems are given to illustrate the point. One is a definite improvement, the other is in my opinion almost a retrograde step.

The first case involves retrofitting of a radio package for a light observation helicopter to include six radios. This is shown in Figure 7. The exact details of this controller have been detailed in a previous AGARD paper (AGARD Guidance and Control Panel Proceedings Number 329, Paper 19; Communications Management - A Vital Link). An operational pilot who tried the system commented that it took ten minutes of instruction to learn how to use it. It is simple, it looks like a radio and it is easy to change fixed and preset frequencies with this system. The drawbacks are that it only displays the frequencies of the radios selected for transmission on each side, (given that panel space was limited this was an acceptable shortcoming), and that it requires hands off the flying controls to change frequencies or radios.

The second system, shown in Figure 8, was admittedly from a prototype anti-submarine warfare helicopter, however, that does not mitigate the bad features of its design. It was designed for a single pilot operation aircraft, with two rear cabin crew, and each station featured 18 pushbuttons and 14 rotary volume controls. The rationale for the proliferation of pushbuttons was that they were inherently more reliable than rotary switches.

The display was blank until a pushbutton had been pressed. The logic was arranged so that the first press of the button brought the set to receive, the second push placed it in transmit/receive and the third push turned the radio off again. Changing transmitters took a minimum of three pushes of buttons, if it was done correctly the first time. After all of that the controller did nothing more than select radios for transmitting and adjust volumes. Changing frequencies still had to be done through a Controller Display Unit (CDU) and took a minimum of five further steps. If the CDU happened to be on another function the pilot was not able to see what frequency he was transmitting on. This second system clearly demonstrates many things, but most important, it shows that a minor technical decision (use pushbuttons as they are more reliable) can have a tremendous impact on the usefulness of the system. The operational requirement was subordinated to the technological requirement.

UPDATING THE MATURE HELICOPTER

Putting together a package of advanced technology is not easy, especially when starting from scratch on a new design. There is always the demand to put a quart in a pint pot and demonstrate improvements in performance, fuel consumption, noise, maintenance and the whole host of other measures of a product. Rather than pursue this avenue, I will attempt to show how a retrofit to an existing helicopter can significantly improve its safety and capabilities, while reducing pilot workload. The devices chosen for the mock retrofit are existing technologies and represent a minimum amount of risk. The helicopter chosen is the venerable UH-1H.

No changes to the engine, airframe or rotor are considered as they are independent of the electronics and in any case represent only a minor change in the existing order of things. Items that are changed are those that will reduce pilot workload while improving capabilities. In addition, no attempt will be made to include such items as MLS or tactical approach aids.

To assist in flying the helicopter a simple AFCS should be installed. It should include an attitude hold feature in pitch, roll and yaw, and permit rate damping when manoeuvering. Each lane should be deselectable, i.e. a malfunction in the pitch channel should still permit the roll and yaw channel to operate. To maintain simplicity, it should not feature any altitude hold, turn to heading or other modes.

Performance monitoring and computation would be greatly enhanced by the addition of a computer, as previously described. This would require inputs from the engine, transmission, the low airspeed sensor, the radar altimeter, the cargo hook and the Doppler navigation system. With the appropriate information in its memory, it would be able to determine if power required was likely to exceed power available, or if any control margins were being approached. It should of course be capable of computations well beyond the normal published limits to cater for the wartime case. Figure 9 shows basically how this could be integrated.

To assist in navigation, a Doppler navigation system should be added, however, it should have a limited number of waypoints, and simple controls. It should require only battery power to insert information, and have a suitable instrument panel display of groundspeed, drift, and at low speed, a set of cross hairs for hovering. The display should be to the right of the radar altimeter

If not already fitted, a radar altimeter should be installed, and situated immediately to the right of the attitude indicator.

The radios should be replaced to minimize the work required to change not only frequencies, but also transmitters. This should be possible with hands on the flying controls. The flying control grips may require slight modification to permit this. The frequency currently being transmitted on should be displayed in front of each pilot on the instrument panel.

Finally, the pilots seats should be made comfortable enough to permit eight hours of flying to be done each day without crippling the pilot.

SUMMARY

In summary, the lessons of previous conflicts have taught us many things about the helicopter and its shortcomings. Many of these lessons have been absorbed, and advances made, however, the electronic explosion presents an opportunity to correct many of the recurring themes that may have slipped through the cracks. The danger lies in letting technology do everything, when a man is capable of performing many functions with equal ease. Future helicopters must not be technological showpieces just because technology has the capabilities, but should have only sufficient technology to do their mission, driven by a firm operational requirement.

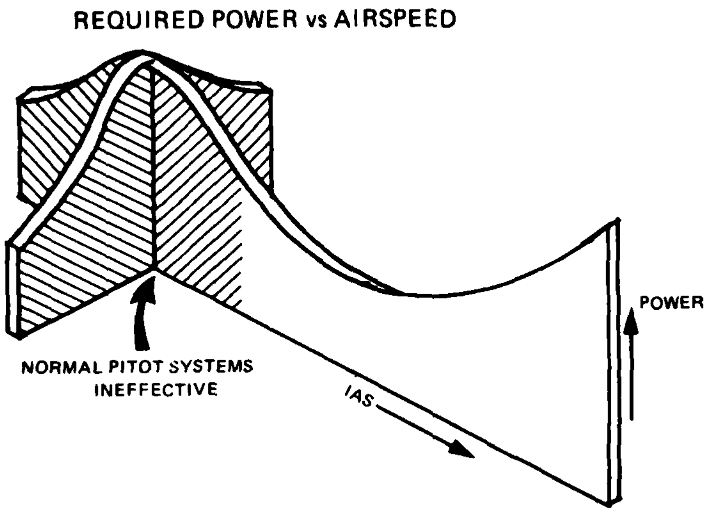


Figure 1. Power Required Versus Airspeed.

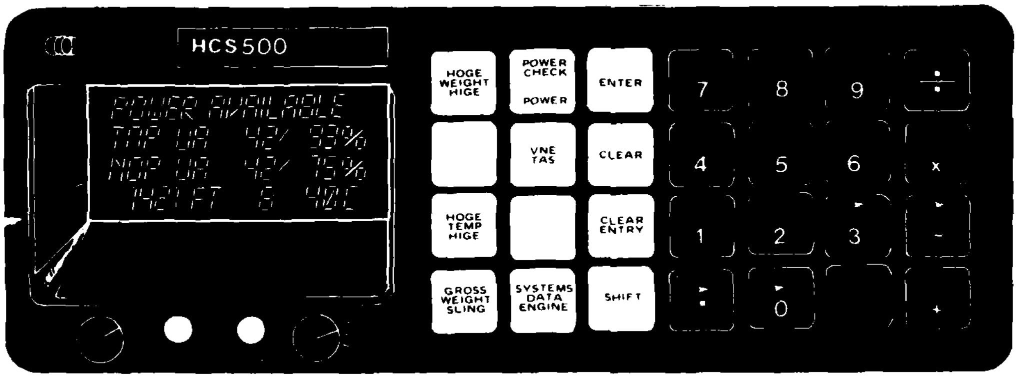
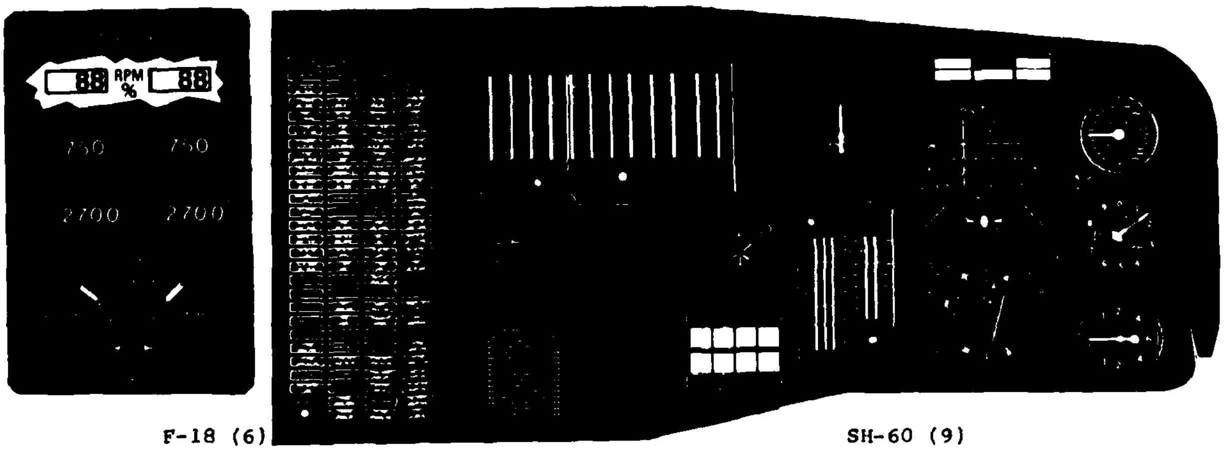


Figure 2. A Typical Helicopter Performance Computer.



F-18 (6) SH-60 (9)

Figure 3. F-18 and SH-60 Engine Instruments.

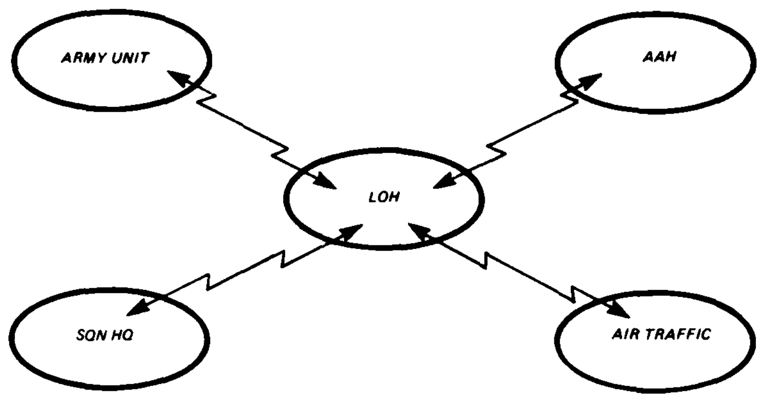


Figure 4. Typical Radio Links.

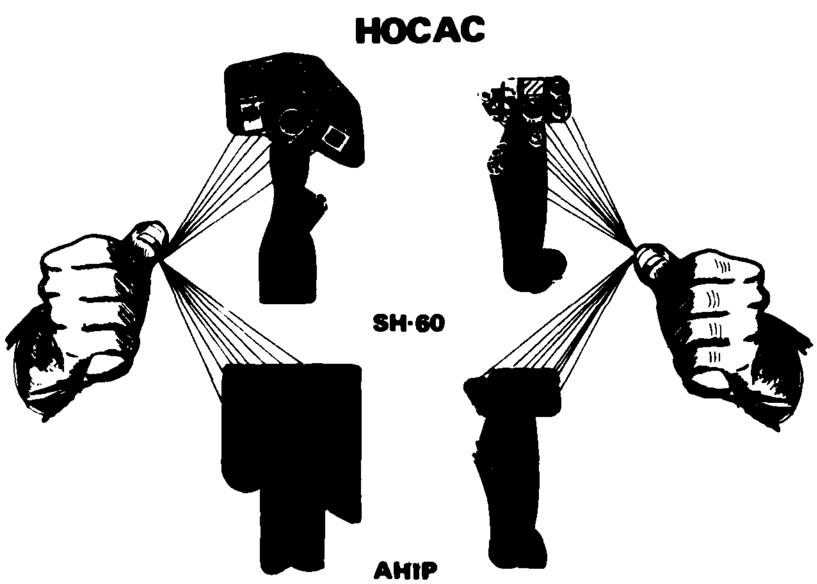


Figure 5. Helicopter Hands On Cyclic and Collective (HOCAC).

Figure 6. F-18 Hands On Throttle and Stick (HOTAS).

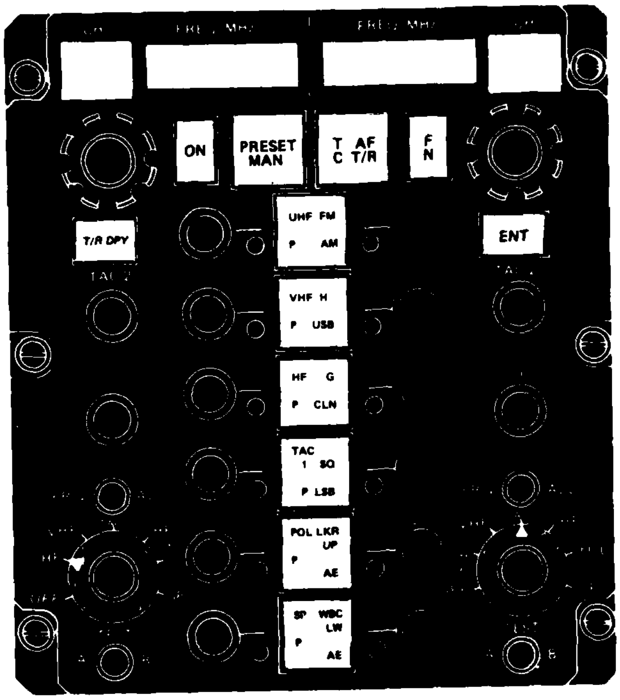


Figure 7. Six-In-One Radio Controller Prototype.

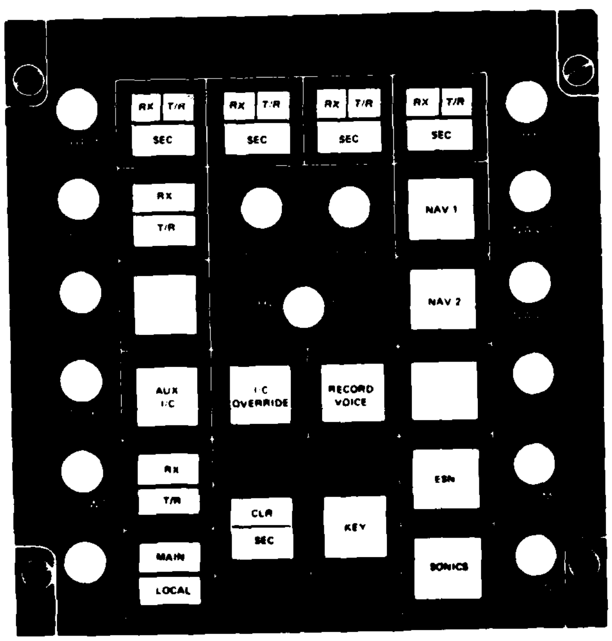


Figure 8. Intercom Controller for Prototype Helicopter.

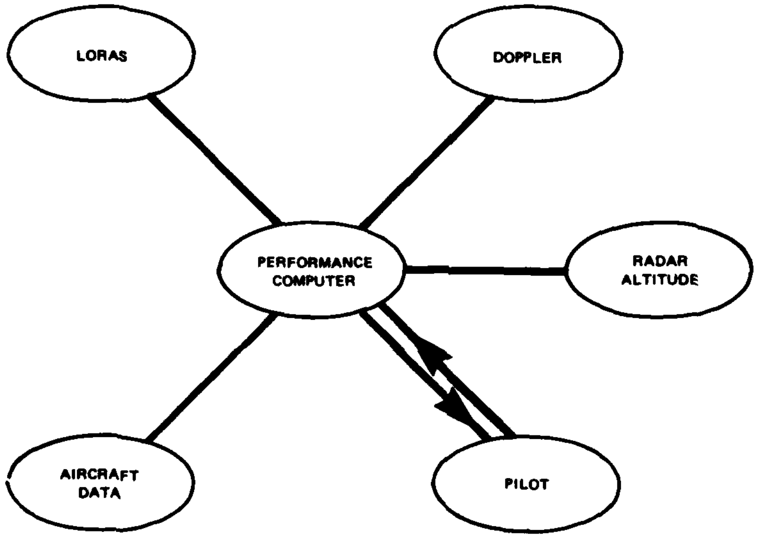


Figure 9. Performance Computer Interfaces.

AUTOMATIC BRIEFING AND MISSION MANAGEMENT SYSTEMS FOR BATTLEFIELD SUPPORT

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SUMMARY

Mission planning and briefing systems have been used for almost a decade to make the best use of advanced avionics fitted to high performance fixed-wing aircraft. As similar equipment is fitted to helicopters to enable them to perform on the battlefield the need for mission briefing systems will become apparent. The paper describes how such a system might be used in the context of a hypothetical future operation and in doing so seeks to highlight some of the technical problems which will be encountered in developing and fielding such equipment. It considers the interaction of the databases which could be exploited by such a system. Lastly the paper considers whether the equipments in service today are developing towards the future systems described above.

INTRODUCTION

Briefing and Mission Management systems for battlefield helicopters are still in their infancy. This is partly due to the relatively simple avionics which have until now been fitted to these helicopters and partly because the operational necessity of these systems has not existed. Within the foreseeable future, both these situations will change. This paper explores these changes and, in the light of our experiences with fixed wing aircraft, tries to ascertain the likely form and function of future mission management systems, how the anticipated advances in technology will enable them to come about, and what limitations technology will place upon them.

THE SITUATION TODAY

The greatest asset the helicopter of today can offer a field commander is its mobility. The operational roles of helicopters have in the past emphasised the rapid transportation of men and materials with great rapidity and over ground of difficult going. The most aggressive manifestation of this role has been the rapid and semi-clandestine insertion of small packets of men into hostile territory. Nowadays, helicopters offer a more potent role. Their ability to transport firepower has made them an altogether greater asset to a commander. But this in itself has caused a limitation of employment - so important are they to his tank-killing capability that a commander deploying limited numbers of helicopters in the defensive battle is forced to keep them back, ready to unleash in the counterpenetration role in order to stop the enemy breaking through his positions.

To date, the greatest asset to crews in achieving their high mobility task has been an improved navigation system. While this has helped operations, particularly at night, the principle navigation tool is still the hand-held map, marked to show troop dispositions as well as terrain and the mission plan, a line drawn across the map, marked off in minutes. Position displays are still rudimentary. Weapon aiming sensors and their appropriate displays are improving. These reflect the fact that hitherto commanders have been unable to rely on the availability of their helicopters in adverse conditions. The extra killing power of the helicopter has forced the requirement for them to be available in all conditions. New sensors for target acquisition and their associated displays are now being fitted. These are the first steps towards a full helicopter avionic fit.

FUTURE DEVELOPMENTS

Whilst the roles of future helicopters may not be extended, their ability to perform these roles will be. They will have to fly aggressive sorties at night or in bad weather often within sight and range of enemy positions. The improved organic air defence capability of a potential enemy will increase and the helicopter will become more vulnerable. As it is not an inherently survivable machine, it relies for its survival on concealment. Concealment at speed is extremely difficult - in some instances any movement will give away position - and requires very low flying to make best use of contours. In fire positions concealment means lurking behind cover but in front of high ground to avoid being silhouetted. Here, modern technology can reduce the risk of discovery through the deployment of fire-and-forget weapons.

So the tasks of modern technology avionic systems will include:

a. To assist in the fast traversing of ground in safety and in adverse conditions.

b. To assist the crew in choosing suitable fire positions.

c. To facilitate flying in the hover.

d. To enable the crew to engage targets at greater ranges and in adverse conditions.

The essence of this is to reduce the burden on the crew of an increasing work-load, and this includes the good presentation of the increased quantity of information which can be used by the crew. This information will include the hazards presented by the terrain - especially important at night - and those presented by forces on the ground. The display should also show the desired route through these hazards, as well as the actual heading and position of the helicopter. The presentation of this information will be possible through the wide range of technologies which are becoming available. These include more precise navigation, giving the crew confidence of their position in relation to the hazards mentioned above, electronic displays, the greater capacity of data storage systems, digital avionics to manage them, and, of course, new sensors.

Much of the technology which will be required for these avionic systems has been driven by developments for fixed wing aircraft. These systems have generated the requirement for the mission planners which are in service today to make the best use of on-board systems. There are parallels which can be drawn between fixed wing close air support and aggressive helicopter operations, but a number of factors change the emphasis:

a. Much more detailed planning with regard to terrain is required for nap of the earth operations;

b. Good fire positions will need to be found to engage the enemy from cover.

c. A greater knowledge of the tactical situation will be required not only to avoid engaging friendly forces and to find the enemy, but also to enable flight commanders to predict the course of the battle and anticipate likely future tasks.

The crucial point is that the battlefield helicopter force must be a fully integrated part of a combined arms battle formation, working closely with the other arms. It will therefore have full access to the formation's command and control network and this will help the flight commander in his task. How might this be used in the mission planning process of the future?

PLANNING AN OPERATION OF THE FUTURE

A field formation has deployed for war, and its aviation elements are dispersed into hides. Each flight headquarters is equipped with a mission planning equipment. The headquarters is fully integrated into the formation's Command and Control (C^2) System and the mission planner is able automatically to update its tactical information store from this source. It already holds a terrain data base for the area, and has been programmed with the performance characteristics of the aircraft. The machine is equipped with rudimentary artificial intelligence which enables it to deduce some likely future tasks, based upon its knowledge of the ground and the tactical situation.

The flight under consideration receives a Warning Order; it is to carry out a counter penetration task on an enemy armoured thrust which threatens to outflank friendly ground forces. Whilst the flight is being prepared with fuel and ammunition, the mission planner carries out an analysis of the task. This analysis covers a wide field - it assesses the likely rate of advance of the enemy armour along its axis and compares this with the likely arrival time of the flight's helicopters. As part of this analysis, it considers the routes and flight times to the areas as well as constraints contained in the Warning Order. It will present the flight commander with a number of areas which it considers suitable for the attack, both from the flank and the front. Should the flight's deployment be delayed, the number of options will reduce. The machine will also take into account the endurance of the flight, balancing approach time with engagement time.

It will be difficult for such a machine to recommend an optimum engagement area. As regards the chance of success in terms of a battle between the helicopter flight and the enemy armour, it is probably best to leave the attack as late as possible. The flight's endurance will be greatest, more time will be available for preparation and the selection of fire positions, and it is likely that more information will be available on the strength and intentions of the enemy column. However, the aim of the engagement is to minimise the damage to friendly ground forces. The longer the engagement is delayed, the less will be the chance of achieving this aim. The flight commander will have to decide where to carry out his attack within the constraints of his orders and balancing the above factors. He can then issue his Warning Order to his crews.

Once the flight commander has selected his primary and secondary engagement areas, the mission planner will present for each area a number of fire positions. For each one it will display possible arcs of fire and maximum and minimum ranges at a variety of heights above ground level. The commander will be able to allocate these positions to the aircraft under his command and adjust the arcs to suit his plan, so that the machine will present to each crew exactly where his commander wishes him to be and what he is to do.

There is a word of caution to be sounded here. The terrain data base may be able to take into account the height of trees and to some extent might carry out some seasonal adjustment. It cannot substitute for detailed reconnaissance, but, when time is short, it can indicate likely suitable areas. Adjustments of a particular helicopter's fire position once it has arrived is likely to be the rule rather than the exception.

The flight commander now ties down the final details of his plan, choosing from the options the route into the battle area, perhaps with a final rendezvous (RV) point before entering fire positions. His plan will include an alternative route with an alternative RV. He will allocate withdrawal routes to a concentration area before deployment to a secondary engagement area. At the end of the battle, each helicopter will withdraw along allocated routes through another RV to either a replenishment point or another hide, again on routes chosen from options presented by the mission planner. Just as a commander does at present, the commander of this flight will try to take into account as many eventualities as he can. Although he will know he can adjust his plan from the ground he will wish to make his aircraft as autonomous as possible to minimise the use of the data link and reduce the effect of its possible loss.

Having completed his plan, the commander is free to attend to his personal administration. The mission planner will revert to its usual operator, possibly the flight second-in-command. It will continue to monitor the situation as received from the formation's command and control system. Should anything happen which might affect the commander's plan, it will alert the operator to the change. When confirmatory orders and timings are received, the commander can confirm this plan or modify it as necessary. He demands print-outs of the area from the digital data store, complete with a situation overlay and routes of his plan. He can give one to each of his crews and if time permits talk them through his plan. The mission planning system will meanwhile load the same information into a portable data store for each aircraft which the crew will use to load the aircraft mission store. This information will be displayed during the execution of the mission so that with the aid of the navigation system, each pilot will be able to compare his actual heading, position, and height with the plan.

The processes involved are represented in Figure 1. As can be seen, the mission planning system makes use of information from its own terrain data base and that of the formation's C^2 data base. It provides an aid to the flight commander who can quickly make decisions based on the information laid before him. It will prompt him to ensure that his plan is complete. It will also fill in relevant extracts from Standing Operating Procedures. It will then provide him with briefing aids in the form of hard-copy orders and printed plans. It will also transfer the digital information to the aircraft, via a Portable Data Store (PODS).

OTHER TASKS

The above scenario is very much concerned with the planning of an operation. But a mission management system should be concerned with other tasks as well. It should be able to organise other aspects of the on-board avionics. The communications system can be presented with instructions for the correct frequencies and codes and voice recognition codes may also be required. Weapons management stores can be organised for safe and effective delivery. Through the formation command net, it will warn friendly forces of the flight's route and operations, arranging safe lanes through which to fly. It can warn the logistic chain of combat supply demands - fuel and ammunition -and the flight time remaining before servicing and undertake the analysis of maintenance data recorded in-flight on the PODS.

INTERACTION WITH AIRBORNE SYSTEMS

The systems within the helicopter which are expected to interact via a databus with the information supplied from the Portable Data Store are shown in figure 2. The navigation system will wish to compare the actual route with the planned route so that it can provide corrections. The displays suite will also make important use of stored information, showing the terrain with planned routes and tactical symbols overlaid on the head down display. The weapon management system may require information of ammunition loaded, and the status of weapon stores would also be displayed. The communications system would also require information from the PODS.

TECHNICAL CONSIDERATIONS

The technical operations required include:

a. Interaction with the formation's C^2 system;

b. Storage of and access to a terrain data base;

c. Analysis of the tactical situation (including threats) to provide a breakdown of likely future tasks (currently done by commanders);

d. Interpretation of formatted orders;

e. Analysis of tasks and comparison with the terrain to produce route options, and fire positions;

f. Presentation on displays;

g. Easy interaction with the operator;

h. Interaction with aircraft either via a portable data store or other data link;

i. Advanced printers.

Central to these operations will be the interaction of databases. There would be the terrain database, stored within the system. The tactical database, containing positions and strengths of friendly forces, known and suspected strengths, positions, and intentions of enemy forces, and other information (artillery fire plans, engineer obstacle plans) would be continuously consulted. Not yet mentioned is the on-board data base held in the aircraft, and this may be dependent on the data link with the ground. The helicopter database will be loaded before the mission by the mission management system either via a portable data store or a wire link. It would contain all the information required for the mission including a large enough terrain database overlaid with tactical symbology to cope with a diversion should the mission plan change in flight.

Amendments to the loaded mission plan may be crucial. The safest course is to preplan alternative routes and tasks as part of the mission plan. But if the mission management system is to be used during the mission, a rapid data link is required. This would give the flight commander the facility of consulting the system should any change of plan be required. For this to be satisfactory, the data link must be secure and reliable, even allowing for a helicopter flying fast and very low, using the ground for cover, in forward areas.

The use of Portable Data Stores (PODS) has been mentioned as a medium for carrying data to the aircraft. Whilst this is only a suggested method, it is one which is proving satisfactory today. PODS are compact, robust and fast. They eliminate the need for wires trailing to aircraft. They have the added advantage that they can be carried during the mission and used to record data for later analysis. A hand-held PODS using EEPROM memory can presently store over half a Megabit of information. Within the timescale this will improve dramatically.

The size of data stores and speed of processing are critical factors today but it is unlikely that this will be so in a few years time. The size of the terrain database is very much a matter for debate at present, and estimates of its requirements vary from large (1 Mbits) to enormous (35 Mbits) for a 1:50 k scale map sheet. This uncertainty is due to the lack of definition of what information will actually be required for storage. There exists a general agreement that contours will be required and that post office symbols will not but there is a large grey area between these extremes. This definition will be an important area of study.

The analysis of tactical information is carried out today by commanders using a combination of common sense, knowledge (or lack of it) of the enemy's modus operandi or his own commander's methods and intuition. With added luck, the good commander does this well. There is a possibility that a machine equipped with some form of artificial intelligence will be able to undertake this task but the results would be rudimentary or merely an extrapolation of events in the past. I suggest that some operational analysis will be required to demonstrate whether this facility is really worthwile.

The analysis of terrain particularly for finding optimum routes is a different matter. The algorithms for this analysis have been worked out, but as far as is known no in-service equipment makes use of them today. This is because the software still needs refining, and, particularly, the machines required to process it are larger than those presently fielded and the time required by them to produce the analysis is too great. All these factors will change within the next few years as processors become faster and more compact, and so fielding the route analysis task will be quite feasible.

The provision of advanced displays and fast printers is an obvious one and the technology of these equipments is being driven hard by many other applications. The interaction with the operator is, however, an interesting area. Mission planning systems are supposed to speed up a rather tedious process, but in the past they have not always achieved this with great success. Our advances in this area seem to owe as much to the light of experience (trial and error) as to the application of common sense or psychology. Some basic rules can be applied:

a. Simple instructions;

b. Use of menus;

c. Diagrams rather than alphanumerics.

One other point is that the NATO standard symbols, agreed after so much discussion, appear to be unsatisfactory on colour electronic displays and will need to be changed.

THE SYSTEMS OF TODAY

Most of our experience to date has been gained with systems for use with fixed wing aircraft. However, Ferranti Aircraft Equipment Department, in conjunction with the Royal Aircraft Establishment have developed the Battlefield Mission Management System specifically for use with helicopters. At first sight, the systems of today seem a long way from the hypothetical system described above. Closer inspection, though, reveals early approaches in the right directions.

One of the first lessons learnt was how necessary the portable data store is, and strangely this is a lesson which seems to have been lost on some of our competitors. The Autoplan system designed to work with the Jaguar NAVWASS system was designed with an early PODS. This was not taken up as part of the system, so that once the pilot had plotted out his route, he had to carry a printout to the aircraft and punch in the data. This was unsatisfactory.

A development of this system for the Tornado, uses a tape cassette (which plugs into the Cockpit Voice Recorder) as the data transfer medium, so that the name of the system is Cassette Preparation Ground Station. As terrain analysis is still carried out by the crew looking at the map, the system uses a plotting table to lift the route plan and other information from the map. The processor carries out some simple calculations relevant to the route - particularly to do with fuel requirements and timings -but the capabilities of the processor are considerably under-used. The machine is equipped with a floppy disc data store interface which until now has been mainly used for storing routine flights or pre-planned missions with some supporting intelligence data. As well as a printer and visual display unit, the machine is fitted with a MODEM interface so that it can be used with a data link.

Our later systems are based upon this form, but have shown steady step-by-step developments. The use of higher level languages made the software more flexible. The displays are improving -in particular, graphical displays show up planning errors more effectively than alphanumerics and, as an alternative to keyboards, we use a grid overlay, to make the systems easier to use. And of course we have gone over completely to solid state Portable Data Stores.

The emphasis of the Battlefield Mission Management System is rather different, reflecting the different requirements of the helicopter. The essential elements of the system are similar -preparation of information for loading into an airborne store via a Portable Data Store - but the information is more concerned with the tactical disposition of forces over an outline map than with route planning. The system has to use a dedicated display as none is presently fitted and the crew can call up a series of formats as required.

CONCLUSION

This paper has described a hypothetical mission management system of the future based on likely requirements, the technology which is expected to become available, and current trends. It has identified technological problem areas which will need looking at and include:

a. Analysis of the desirability of a forward data link into the flying aircraft;

b. Better definition of the terrain data base;

c. Assessment of artificial intelligence capabilities for tactical analysis;

d. Improved methods for operator interaction.

One aspect which has not been covered in this paper is perhaps the greatest limiting factor of all - the user's perception of what these systems have to offer. In the fixed wing world, where the use of mission planners is now accepted, initial specifications always emphasise a minimum solution to data entry and yet after discussion this requirement is expanded. It is hoped that this paper has at least given an insight into future possibilities and will act as a stimulus to discussion.

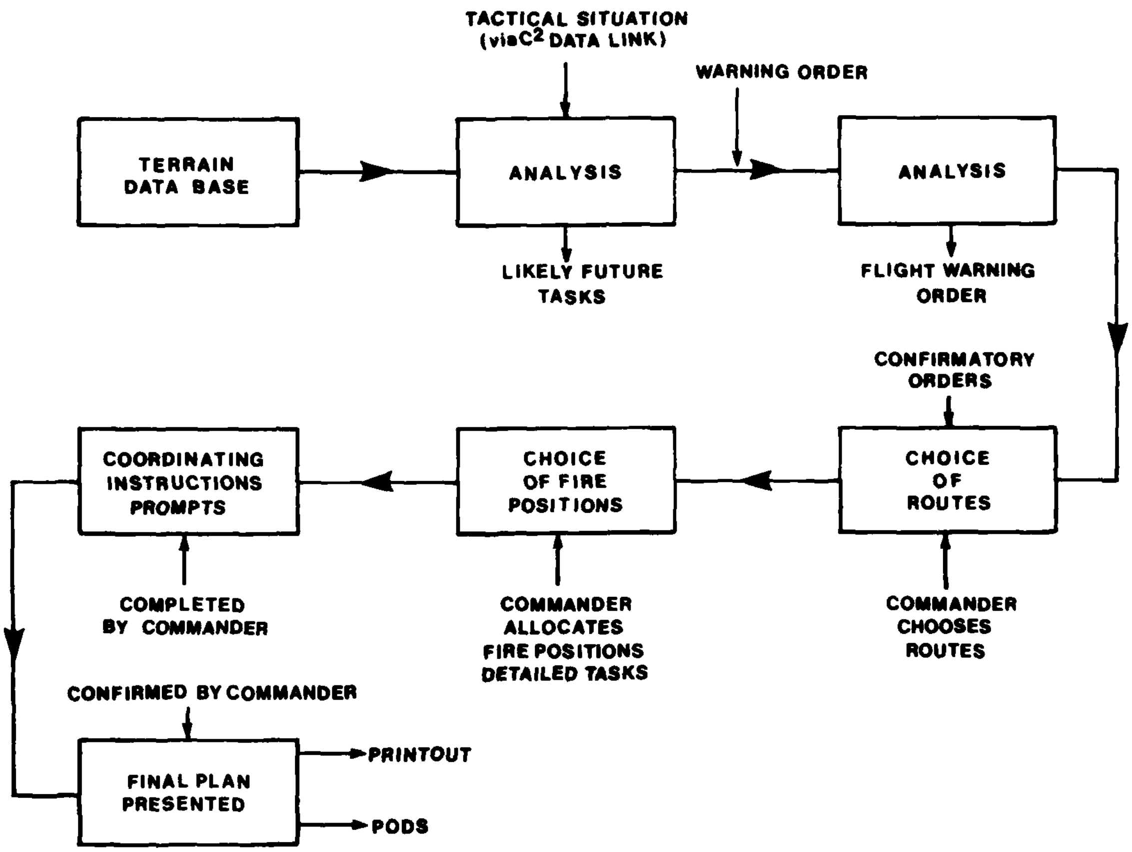

FIGURE 1 THE PLANNING PROCESS

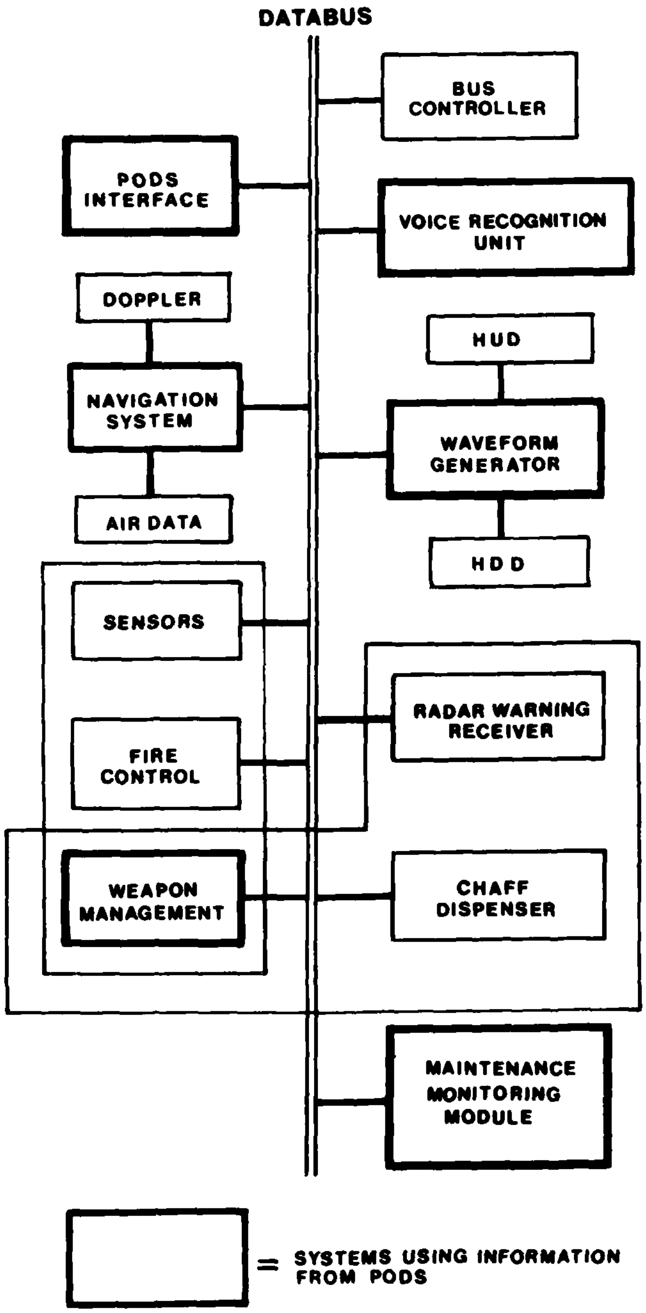

FIGURE 2 AVIONIC SYSTEMS INTERACTING WITH PODS

A COMPUTER-GENERATED TOPOGRAPHIC DISPLAY SYSTEM FOR NAVIGATION AND MISSION PLANNING IN HELICOPTERS

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SUMMARY

The Integrated Mission-Planning Station (IMPS), under development by the U.S. Army Avionics R&D Activity, is a computer-generated topographic display system. The IMPS will provide not only a greatly enhanced navigation capability, but also a combination of dramatic improvments in cartographic support, map information content, and aviator-map interactions. The powerful computational capability of the IMPS can be used to present a shaded "relief map," to display areas masked from visual or radar observation, to construct perspective views of key terrain, to support a self-contained terrain correlation navigation system, and to permit rapid solutions to problems of previously forbidding complexity.

This paper briefly describes the tasks conducted to provide detailed human factors engineering specifications for the construction and programming. The outcome of these efforts is described in terms of an overview of system components and functions. Each of the major functions of the IMPS system is discussed, and the operational requirements, present deficiencies, and IMPS capabilities are identified.

INTRODUCTION

BACKGROUND

Army Aviation Tasks

Only a few years ago, the Army used helicopters for little more than medical evacuation. Today helicopters are an integral component of the combined arms team and perform a great variety of tasks. Army aviators must be prepared to enhance the ground commander's capabilities in one or more of the five functions of land combat: firepower, mobility, intelligence, command and control, and combat service support.

These functions require the performance of an impressive array of diverse missions. The missions required depend upon the aviation unit type, the combat arm that the aviation unit serves, the aircraft type, and the specific tactical situation. Thus, the potential missions of Army aviation are countless. A common theme in nearly all of these missions, however, is the requirement for terrain flight.

Terrain Flight

Terrain flight is the tactic of degrading the enemy's ability to detect the aircraft by using landforms and vegetation for cover and concealment. This tactic requires flight close to the earth's surface and includes low-level, contour, and nap-of-the-earth (NOE) techniques. Low-level flight generally employs a constant heading, altitude, and airspeed. Contour flight is conducted in very close proximity to the ground and requires altitude changes to conform to the contour of the earth, while maintaining a generally constant heading. NOE flight is flight as close to the earth's surface as vegetation and obstacles will permit, varying course, airspeed, and altitude in order to take maximum advantage of the cover and concealment offered by terrain, vegetation, and man-made features. In practice, the aviator may use a combination of these three techniques during a single mission. The most important determinants of flight techniques are the enemy situation and the availability of masking terrain. It is critical that the aviator be aware of the positions and altitudes at which masking is or is not available. Flying unmasked sharply reduces survival probability in the high-threat environment. On the other hand, unnecessary NOE flight is inefficient because more sorties can be flown or greater distances covered in a given period using contour or low-level flight. Furthermore, high altitudes offer a greater margin of safety in dealing with aircraft emergencies and hazard avoidance.

Geographic Orientation

Whatever the combat task or flight technique, the Army aviator must be able to maintain his geographic orientation at all times. Aviators must be able to plan and execute their missions precisely in both time and space and relate their momentary position to their planned route and to the movements of other friendly ground and air forces. In the critical timing of events on the modern battlefield, disorientation is tantamount to mission failure. As a minimum standard, Army aviators are expected to navigate to an accuracy of 100 meters at all times (1). Furthermore, aviators are expected to navigate in unfamiliar terrain, around the clock, and in adverse weather conditions.

Descent to NOE flight levels greatly increases the likelihood of geographic disorientation due to the aviator's limited view of checkpoint features useful in navigation. While NOE flight serves to mask the enemy's view of the helicopter, it often masks the aviator's view of potential checkpoints. The view of the surrounding terrain may be limited to features within 100 meters of the aircraft. Features often cannot be seen in their entirety, and the extremely low angle of view increases the difficulty of determining the contours of visible landforms.

Both anecdotal evidence and controlled field tests have indicated that the percentage of NOE sorties in which the aviators experience no navigation problems and remain well oriented throughout the flight is exceedingly small. In an early study by Thomas (2), aviators were able to navigate a 6-km course only 48% of the time without becoming disoriented. In a Canadian test (3), aviators were able to reach the designated end points of short courses only 61% of the time. In a more recent experiment (4), major course deviations (500-2000 meters) occurred in half of the 12 30-km course attempts by Cobras, and only two crews navigated the entire course without missing checkpoints, circling, doubling back, or experiencing other navigational difficulties. In another recent test (5), 35 Army aviators flew NOE missions in a series of 279 test flights. The results of this experiment indicated that the probability of successfully acquiring both the initial point (IP) and a subsequent landing zone (LZ) was .65.

Mission Planning

The results of Anacapa laboratory studies of the accuracy of geographic orientation by Army aviators (6, 7) are in agreement with the field tests, and suggest that the navigation problems are rooted in the difficulty of the map interpretation and terrain analysis tasks. Such findings suggest that the aviators must also experience great difficulty in the extensive mission-planning tasks required for each mission. The successful accomplishment of many of these tasks depends heavily upon the aviator's ability to extract voluminous information from maps. For example, the aviator must study and visualize the overall situation and topography; select engagement points, observation points, or landing zones; determine primary and alternate (masked) routes of flight; select air control points, checkpoints, and barrier features; and determine flight modes, altitudes, speeds, and durations. Each of these activities places an onerous information compilation and processing burden upon the aviator, and omissions or errors might well prove to be disastrous. Although previous map display systems have been designed primarily for assisting in navigation, it would appear that the potential contribution of a map display might well be greatest in aiding the performance of mission-planning and tactical decision-making tasks.

A COMPUTER-GENERATED SYSTEM

The Integrated Mission-Planning Station (IMPS) computer-generated topographic display system will provide not only an enhanced navigation capability, but also a combination of dramatic improvements in cartographic support, map information content, map-orientation computations, and aviator-map interactions. Some of the potential advantages of the IMPS system are discussed below.

The single most important advantage of the IMPS system over paper maps or projected maps displays is its potential for truly comprehensive and rapid response cartographic support. NOE flight requires the use of large-scale maps (1:50,000 or larger). The smaller scale maps (such as 1:250,000 and 1:500,000), which are designed for conventional flight, do not portray sufficient detail for NOE navigation. Only a very small percentage of the earth's surface is currently mapped in large scale, and in the event that a conflict arose in an unmapped area, it could take months to develop conventional topographic maps. Even photo-base maps could require weeks for preparation. In contrast, it is feasible to obtain the data required to support computer-generated display systems in a matter of hours.

A second advantage of the IMPS system is its capability for providing operator control of the content of the displayed information. Many of the problems in using contemporary 1:50,000-scale topographic maps stem from the fact that maps were designed to fulfill the requirements of all branches of the Army. The result is a compromise product that is densely packed with data but is not optimal for any single user. The Army aviator, because of his variety of roles, may need many different types of information on different missions or in different phases of a single mission, yet map clutter must be avoided to the greatest extent possible. Aviators using a computer-generated system can select the information that is needed to provide a map optimal for the momentary situation. Aviators can control the classes of information that are displayed (vegetation, hydrography, etc.) and the specific features of a given class to be portrayed (deciduous trees, perennial streams, etc.) In addition, map scale and contour interval can be changed at will to tailor the map to the aviator's changing requirements.

A third advantage of the IMPS over conventional map products is its powerful computational capability. For example, the IMPS can be used to:

- Show the general lay of the land by use of shaded elevation bands to indicate high and low areas
- Present a shaded "relief map" enhanced by contour lines
- Display the areas masked from visual or radar observation given known or likely enemy positions
- Construct oblique, perspective views of terrain to familiarize the aviator with the landforms as they will be seen during the mission
- Perform navigational computations pertaining to airspeed, elapsed time, or wind vector considerations over a given flight route
- Interact with a terrain correlation navigation system similar to that used in the cruise missile, which is small, lightweight, accurate in all weather, self-contained, and essentially invulnerable to countermeasures

A fourth advantage of the IMPS is that it offers a truly interactive system. An aviator can enter information such as map annotations, coordinates of objectives, planned routes, and so forth. These items of information can then be selected at will, and used in the computations described above. Furthermore, the "intelligent" nature of the system can permit its interrogation by the aviator to determine certain characteristics of the portrayed features, such as tree height and crown cover. The interactive nature of the IMPS can remove some of the natural limits to the aviator's decision-making capabilities and permit him to rapidly solve problems of previously unthinkable complexity.

PROJECT APPROACH

The approach employed to meet the project objectives included a series of overlapping and iterative information-gathering and system design activities. Documents related to Army aviation tasks were reviewed to identify functional requirements for the IMPS system. Aviators were observed during the conduct of mission-planning activities performed to meet the requirements of hypothetical missions selected by the author. Aviators were also observed performing navigation tasks during NOE flight in a variety of terrain types.

Formal interviews were scheduled with aviators of the 101st Division to aid in determining the optimal aviator-computer interactions for the processing, storage, and display of topographic and tactical data. Additionally, interviews took place with aviators of the 2nd Armored Division, the 6th Cavalry Brigade, and the Career Training Division of the Army Aviation Center, in order to evaluate specific characteristics of the preliminary conceptual designs. During these interviews, mission scenarios and simulated control and display surfaces were employed to graphically portray the capabilities of the system and encourage comments and suggestions from the aviators.

Aviators of the 101st Division completed a questionnaire designed to identify current practices with paper maps that might influence the IMPS system design, to examine opinions of aviators regarding the basic requirements for such a system, and to determine aviators' assessments of various IMPS special features. Subsequently, over 100 aviators completed an extensive survey by rating the importance of topographic and tactical features used in meeting Army aviation mission requirements (8).

Concurrent investigations at Anacapa Sciences include a basic research program exploring the effects of various display variables on the perception of topographic symbology. Many of the interim research findings have been useful in defining the characteristics of the IMPS displays, as well as the sizes, shapes, and colors of the displayed symbology (e.g., 9, 10, 11).

Mission analyses were developed from baseline scenarios to identify required system tasks. The specific tasks were then analyzed to determine human performance parameters, hardware/software capabilities, and the tactical environmental conditions under which the tasks are conducted. Specific controls, displays, and procedures evolved from the task analysis.

Given the system purpose and the user capabilities, a menu-selection dialogue was judged most appropriate for the IMPS system. Analyses were conducted to ensure that feedback of system status was continuously provided to the operator, that event sequences were consistent from one mode to another, that consistency was tempered by flexibility and that special prompting and assistance messages were provided to aid the operator in the prevention, detection, and correction of errors.

SYSTEM COMPONENTS AND THEIR FUNCTIONS

The IMPS is composed of two major systems: the airborne system and the ground-based system. The ground-based system includes all of the features of the airborne system and has additional capabilities for mission planning and data-base editing.

THE AIRBORNE IMPS COMPONENTS

The aviator interacts directly with four components: the color CRT map display, the magnetic tape loader-copier, the control-display unit (CDU), and a joystick-type control called the airborne hand controller (AHC).

Color Map Display

A multi-color map display is necessary to meet the extensive visual search and feature coding requirements of Army aviation. Reviews of the literature (10, 12) have revealed that color codes are nearly always extremely beneficial in search and identification tasks. This is an important finding because NOE navigation requires continuous orientation through searching for and identifying topographic features on the map, and correlating them with features seen in the real world. The finding that color codes are of increasing value with increasing map complexity (13) is particularly applicable to the densely packed features on a topographic map.

An additional requirement for a color display stems from the resolution limitations of electronic display mediums. The tiny shape codings possible on paper maps (such as the cross-hatched railroad track symbol) are not currently achievable in a pixel-matrix format, and must be replaced by color codes.

Although many emerging display technologies (electro-luminescent, light-emitting diode, plasma, etc.) offer specific advantages, at the present time only the cathode ray tube (CRT) can meet the multi-color map display requirement. The CRT to be used in the aircraft cockpit is assumed to be approximately 6 to 8 inches in width employing a 512 x 512 pixel matrix. Navigation sensor data, augmented by a terrain correlation technique, are used to identify the aircraft's current position on the map display. The map display translates and rotates (in real time) in response to aircraft motion. The aviator has control over a large selection of display formats and capabilities, as described later in this paper.

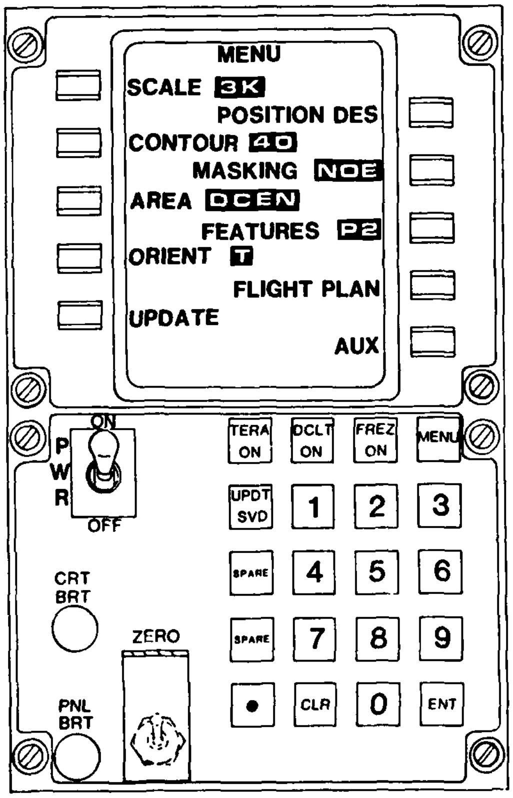

Figure 1. The airborne control-display unit.

Magnetic Tape Loader and Copier

The magnetic tape loader and copier consists of a small electronics unit and removable hermetically-sealed cassettes. The four-track cassette provides storage for elevation data; vegetation, hydrographic, and cultural data; intelligence and operations data; and mission-planning and in-flight annotation data. The geographic area storable on the cassette depends upon the resolution or grid-point interval of the data. The high-resolution data desired for Army aviation use will permit the geographic area stored on a single cassette to be 100 x 100 kilometers--approximately 16 times the area shown on a standard 1:50,000 scale paper map.

The provision of a tape copier greatly simplifies the logistics of map distribution and tactical data exchange. Upon arrival at a new area of operations, the aviator can simply borrow the required cassette and copy the data on a blank or previously used cassette. The distribution of tactical overlay data from higher headquarters can be handled in the same efficient manner.

Airborne Control-Display Unit

The airborne control-display unit (CDU), shown in Figure 1, is an integrated, multi-purpose module which provides the aviator with his primary means of exercising the extraordinary flexibility built into the IMPS system. The uppor portion of the CDU is composed of a small monochrome CRT that displays 12 (16-character) lines of data. The top line is reserved for advisory data, and the bottom line is used as a "scratch pad" to echo keypad entries. The central 10 lines may be used for data display, prompting messages, or labels for the adjacent line select keys. The labels change to indicate the current function of each line select key because the functions of any given line select key will differ depending upon the type of

transaction in progress. The versatility of this display-control concept provides an easily understandable method for leading the user through operational sequences, and it also eliminates the need for hundreds of single-purpose control and display elements.

The lower panel of the CDU provides several different types of controls. On the left side of this panel are the power on-off switch, the CRT and panel brightness knobs, and a "zero" switch to erase secure information in the event of an emergency. The right side of the lower CDU panel is devoted to a numeric keypad and ENTER button for data input, as well as several special-purpose pushbutton controls. The CLEAR button, when pressed once, clears the last digit shown on the scratch pad. A second press clears the scratch pad completely. The MENU button is used to call the display page that presents the function initiation points for all transactions with the IMPS. The LAST PAGE button may be used to display the CDU page presented immediately before the page currently displayed, in order to correct errors or accommodate special strategies for employment of the IMPS dialogue. The other special-purpose buttons, TERRAIN AVOIDANCE, SAVED UPDATE, DECLUTTER, and MAP FREEZE are discussed in subsequent portions of this paper.

Airborne Hand Controller

The airborne hand controller (AHC) is composed of a miniature joystick and three adjacent push buttons. The joystick is used by the aviator to slew the map in order to expand the viewable area, to update the navigation system, and to perform other special tasks. The AHC is also used to move a cursor across the map display in order to designate points in the terrain. In this manner the aviator can obtain range, bearing, coordinate and elevation information regarding the designated points. The cursor may also be used for map annotation and other position-indication tasks.

THE GROUND-BASED IMPS COMPONENTS

The ground-based IMPS components include all of the airborne equipment described above. In addition, the IMPS also provides a pressure-sensitive screen over the color CRT, an alphanumeric keyboard, and map overlay digitizing equipment. The CDU employed by the ground-based system is identical to that employed in the aircraft except that additional software is provided for special IMPS tasks performed at the Tactical Operations Center (TOC).

OPERATION OF THE AIRBORNE SYSTEM

This section of the paper describes the procedures employed in operation of the airborne IMPS system. The procedures are defined for eleven sets of functions performed by the system. The requirements for each set of functions, and the present deficiencies in providing these functions, are briefly described, followed by a short discussion of the capabilities of the IMPS system for meeting the requirements and overcoming the deficiencies.

THE AIRBORNE MENU PAGE

A single CDU page is used for initiating all procedures. This page, called the MENU page, is in Figure 1.

The line labels on the MENU page identify nine sets of related functions performed by the airborne system. Pressing the line select key adjacent to one of the labels initiates the procedural sequence required for performance of that function. An additional two sets of functions, tape copying and flight simulation, can be initiated through the auxiliary (AUX) functions button.

In addition to its role in initiating functions, the MENU page also provides a status display, using reverse video to indicate current IMPS settings pertaining to map scale, contour interval, displayed area, map orientation, masking portrayal, and feature selection rules. The menu page is accessed through the MENU button on the lower portion of the CDU and can be called at any time, regardless of the transaction in progress.

In the subsequent paragraphs, operational procedures are described for the following eleven sets of functions:

Map Scale
Map Contour
Map Area
Map Orientation
Position Update
Position Designation
Masking and Intervisibility
Feature Selection
Flight Plan
Flight Simulation
Tape Copying

MAP SCALE

Requirements. Aviators attempt to obtain maps of several different scales for use in the planning and conduct of missions. Maps in 1:250,000 scale, valued for their wide area coverage, are useful in depicting the overall battlefield situations. Maps in 1:25,000 scale show small areas, but provide fine detail for study of the objective or features along the flight route. Maps in 1:50,000 scale are typically used for navigation at NOE altitudes because they provide the minimum required detail for correlation of map and terrain features.

Present Deficiencies. Although NOE flight requires the use of large-scale maps (1:50,000 or larger), only a small percentage of the earth's surface is currently mapped in large scale. Attempting to use small-scale maps (1:250,000 and 1:500,000) is almost certain to lead to disorientation during NOE flight. Even where large-scale maps are available, however, their use in the cockpit is often awkward because many different map sheets may be required for a single mission.

IMPS Capabilities. The IMPS data base may be employed to portray the terrain in any of four scales selectable by the aviator. During mission planning, small-scale portrayal can be used for overall route selection, alternated with large-scale portrayal for scrutiny of specific terrain features. During mission conduct, the map scale can be changed for greatest utility given the aircraft speed, the view of surrounding terrain, the information necessary for orientation, and the desired look-ahead distance. The size of the data base (100 x 100 km) is independent of map scale, so that "flying off the map" is no longer a drawback to large-scale map use.

MAP CONTOUR

Requirements. For Army aviators, terrain relief--the shape and height of landforms--is probably the most important class of information on a map. Landforms are stable over long periods of time, are often unique in appearance, and are nearly always discernible. These considerations make landforms the primary reference for geographic orientation. Furthermore, terrain relief has great tactical significance for military operations in general, and NOE flight in part ular. It is essential for all types of missions that aviators be able to extract terrain relief information from ntour data presented on maps. Terrain relief is depicted on most topographic maps through the use of contour lines. The contour line technique is the only terrain encodement scheme which meets the severe requirements of NOE navigation: depicting very large elevation ranges while maintaining the precision required for referencing relatively small terrain features.

Present Deficiencies. Unfortunately, the perceptual task of relating contour lines on a map to the terrain relief on the ground is the most difficult aspect of map interpretation during NOE flight. Even experienced Army aviators commonly encounter difficulty in performing contour interpretation (6, 7). This difficulty tends to result in geographic disorientation and limits the ability of aviators to become reoriented.

The ideal contour interval for landform portrayal depends upon the dimensions of the features, the steepness of their slopes, and the precision required by the aviator. A small contour interval is useful for defining relatively small, flat terrain features. The same contour interval cannot be used for steep terrain features because the lines would abut and form dark areas devoid of information. The contour interval on paper maps, however, is fixed. The cartographer must select a compromise interval (although supplementary contours are sometimes added).

IMPS Capabilities. The IMPS system is capable of displaying any contour interval selected by the aviator. The contour interval can be quickly shifted to meet the changing requirements for precision and to improve the ease of contour interpretation. The general lay-of-the-land, or the configuration of specific landforms, may be carefully examined by selection of the appropriate map scale and variation of the contour interval. The IMPS may also be employed to produce extremely precise relief shading without incorporating human errors of interpretation. The shading algorithm developed at AVRADA (14) produces an extremely realistic representation of a three-dimensional landform from elevation data, and will greatly reduce the information processing burden imposed upon the aviator by the contour interpretation task. An example is shown in Figure 2. In addition, the IMPS provides the aviator with an elevation guide capability. Rather than employing a separate small-scale depiction, the entire map area is used to display the gray-shaded contour bands so that the guide may be used in conjunction with data providing precise navigational reference points. Furthermore, the aviator can adjust the guide bands in order to optimize the utility of this feature.

A special case of the elevation guide is the terrain avoidance mode, called by a special purpose button (TERA) on the lower portion of the CDU. The terrain avoidance mode clearly identifies terrain above and below aircraft altitude through the use of a color-coding scheme. The terrain above aircraft altitude will be depicted in amber, and the terrain below will be shown in gray. The only other features shown in this mode are enemy activity and flight hazards (red). The terrain avoidance mode will be of particular importance when the view of the terrain is obscured by smoke or other atmospheric attenuation, or at night when the use of night vision goggles may be inappropriate.

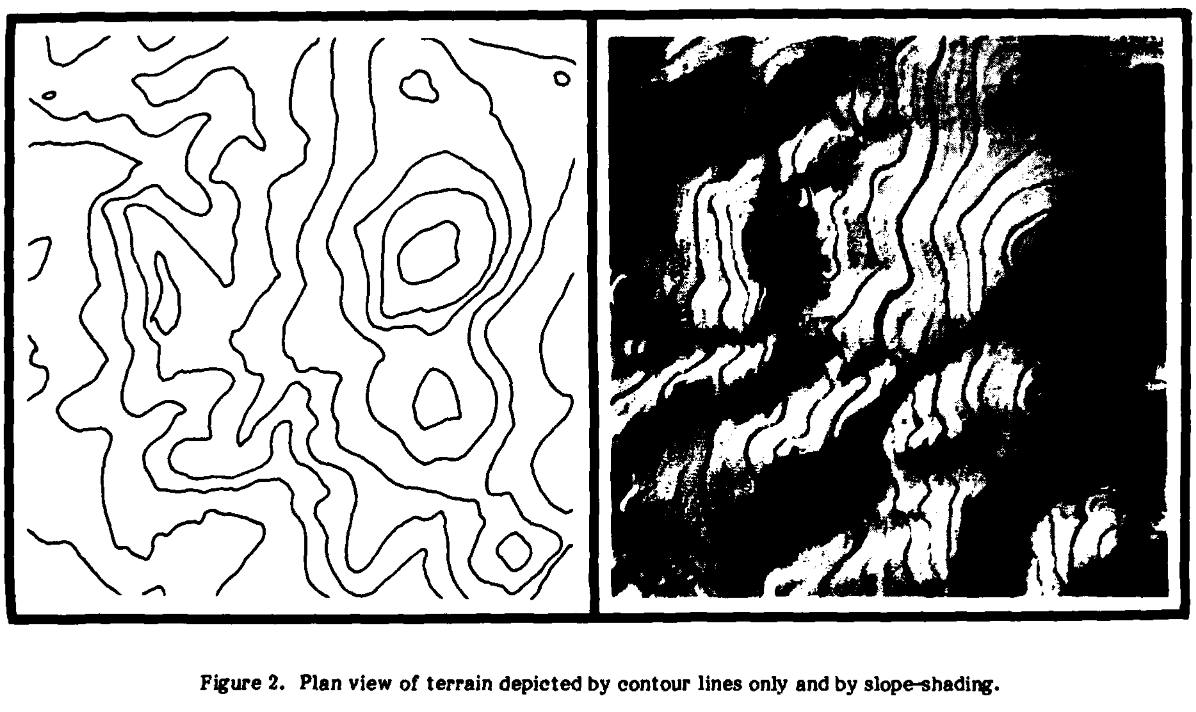

Figure 2. Plan view of terrain depicted by contour lines only and by slope-shading.

MAP AREA

Requirements. The aviator is concerned not only with the position of his aircraft on the map, but with the positions of the surrounding military units, obstacles, checkpoints, and landforms. Although the primary area of interest is usually forward of the aircraft, the aviator must also be able to examine mapped data to the sides and to the rear of the aircraft in order to annotate the map with positions of bypassed targets, downed friendly aircraft, or other tactical data. In addition, it is sometimes necessary to study sites at some distance from the aircraft's present position--for example, the area surrounding the mission objective.

Present Deficiencies. Paper maps provide good flexibility for examination of areas adjacent to, or distant from, the aircraft. In the cockpit, however, the required folding, unfolding, and refolding of maps often becomes awkward.

IMPS Capabilities. The IMPS incorporates several features that enable the aviator to conveniently study portions of the data base near to or distant from the aircraft's present position. First, the aviator may select the location of the aircraft's present position indicator--either centered on the display screen to permit portrayal of the terrain surrounding the aircraft, or decentered to provide maximum look-ahead distance. Second, the IMPS system will display the area surrounding any position in the data base designated by the aviator. Third, the displayed area may be changed by slewing the windowed area to increase the viewing distance in any direction.

MAP ORIENTATION

Requirements. The majority of aviators prefer to orient the map so that the spatial arrangement of the map features is congruent with the arrangement of topographic features on the ground. This map orientation simplifies navigation by terrain referencing because it minimizes left-right confusions. A small proportion of aviators, however, prefer to maintain the map in a constant orientation and mentally transpose the spatial relationships of map and ground features. Still others prefer to turn the map to the cardinal direction nearest the aircraft's planned or actual flight heading.

Present Deficiencies. Although paper maps are easily turned to any orientation, determining their correct orientation can be confusing when the aircraft is following a sinuous course--as is often required for NOE flight. Because all of the alphanumeric information on a map (place names, spot elevations, grid line numbers, etc.) is oriented to be read when the map is north-up, some aviators attempt to alternate between north-up and heading-up map orientations, risking momentary disorientation with each change of map position. Others elect to maintain the north-up mode to reduce this source of confusion, even though the spatial relationships of map and terrain features are not optimal.

IMPS Capabilities. The IMPS system is capable of providing any desired map orientation, and alternating among orientations at will, whether the aircraft is on the ground or in flight. The terrain correlation subsystem continues to function during any map orientation. The map orientation can be set at any cardinal direction up, or at track, heading, course, or ACP up. These settings are, of course, mutually exclusive. The track-up model is based on prior movement over the ground and is essentially a wind-corrected course upward display. In helicopters, a stable track can be computed only when the aircraft is proceeding at some given forward speed, probably about 10 knots (15), so a heading-up capability must also be provided for use during hovering. Heading up is simply the direction the aircraft is currently pointed. Course up is based on an aviator entry of any desired direction. The ACP-up mode sets the top center of the display at the current compass bearing to the selected ACP. A map "freeze" capability is also provided to temporarily halt map motion for the cases in which a stabilized image may aid in examination of map details.

POSITION UPDATE

Requirements. Navigation at NOE altitudes requires continuous maintenance of orientation by identifying terrain features along the route and correlating them with features depicted on the map. Because of the aviator's limited view of the terrain, extremely reliable checkpoints may be available only intermittently. At these checkpoints, the aviator must "update" his estimated position on the map, reducing any accumulated error to a minimum. Although navigation during NOE flight is one of the most demanding tasks ever required of an aviator, there is no substitute for accurate position updating. The most direct method of updating is recognition of specific terrain features on the ground near the aircraft and identification of their referents on the map. In the absence of such correlation, or because flying near recognizable features may be tactically inadvisable, aviators must be able to update their position by knowledge of the range and/or bearings of distant objects.

Present Deficiencies. Both anecdotal evidence and experimental data indicate that the average Army aviator is unable to consistently perform NOE navigation to the required level of accuracy. Furthermore, in scout aircraft, navigation is often the responsibility of a crew chief who is insufficiently trained for this difficult task and may be unable to do more than recognize the most obvious man-made terrain features.

IMPS Capabilities. The IMPS system incorporates the most recent development in automated navigation--terrain correlation technology, similar but superior to that employed by "cruise missile" weapons. The system uses the digital elevation data, the ground clearance (determined by a radar altimeter), and doppler data to continuously update the navigation display. The system will present the aircraft position with great accuracy. The aviator will have only to set the initial position of the aircraft, and (possibly) to make occasional corrections enroute by slewing the map.

POSITION DESIGNATION

Requirements. All military activities are dependent upon the rapid and accurate transmission of geographic data. The determination, recording, and communication of position designations are continually recurring requirements for Army aviators. Examples include handing off targets, calling for artillery support, requesting tactical air strikes, gathering battlefield intelligence data, specifying rendezvous points, coordinating battle team operations, performing resection for navigation checks, and identifying LZs. Positional information is transmitted by simple grid coordinates, coded coordinates, preselected position code names, and range and bearing from known positions. Latitude and longitude is used for communicating positions to supporting tactical aircraft.

Another type of data often required regarding specific geographic positions is that of elevation. A primary use of elevation data is for terrain avoidance during conditions of limited visibility. Elevation data are also useful for navigation by terrain association through provisions of relative heights of groups of features seen in the terrain, and absolute heights of features near the aircraft. In addition, elevation data are useful in increasing the effectiveness of supporting artillery fire.

Position designation often includes the requirement for a graphic record of specific positions of features on the ground. These annotations serve as readily recognizable cues for mission activities, as well as a method of summarizing intelligence data in a concise manner. Although numerous annotations are provided prior to flight, many must be entered in the aircraft during performance of the mission.

Present Deficiencies. Present procedures for position designation are often awkward and inaccurate. For example, although the military grid reference system permits the location of a point within 10 meters, use of the system to this level of accuracy requires that a plastic coordinate scale be overlayed on the map. The recipient of these coordinates also must use the coordinate scale in order to plot the designated point on the map. Army aviators have found that the coordinate scale is unsuitable for use in flight, and almost never attempt to designate positions closer than the nearest 100 meters (six-digit grid coordinates). Even at this level of accuracy, it is commonly acknowledged that errors are made in determining the coordinates, communicating these coordinates, and in applying them to locate a point on the map. Similarly, without the use of a protractor and distance-scaled straight edge, it is difficult to determine or apply

range and bearing information to designation positions on a map. Because these devices are also unsuitable for cockpit use, aviators simply make rough estimates of range and bearing--even in such critical applications as target hand-offs.

The use of the data on 1:50,000-scale topographic maps to convert between grid coordinates and latitude-longitude figures is difficult under any circumstances. In the helicopter cockpit, it is nearly impossible. Determination of elevations at specific positions requires that the aviator count up or down to the nearest index contour line and follow this line to a point where its elevation is given--an awkward and error-prone procedure.

IMPS Capabilities. The IMPS computer is capable of speeding point designation, while improving accuracy and precision. Coordinates and elevation of any location on the display may be determined simply by positioning a cursor; or, keying in coordinates will result in a symbol appearing at the designated position. The designation of positions may also be achieved by entering range and bearing data or preselecting code names. The CDU also has provisions for annotating the map display, converting UTM coordinates to latitude and longitude, and computing range and bearing of positions from any site on the map display.

MASKING AND INTERVISIBILITY

Requirements. The term "masking" refers collectively to cover from weapons fire and concealment from visual, optical, or electronic observation. Masking is the central objective of terrain flight, whether of the NOE, contour, or low-level type. It is critical that the aviator be aware of the positions and altitudes at which masking is available. Helicopters exposed to the enemy for more than a few seconds are likely to be destroyed. However, NOE flight should be avoided when it is safe to do so because more sorties can be flown or greater distances covered using contour or low-level flight. In addition, higher altitudes provide a greater margin of safety in dealing with aircraft emergencies and hazard avoidance.

Masking considerations are crucial not only for the selection of flight altitudes and flight routes, but also for planning radio communications; determining enemy and friendly fields of fire; predicting checkpoint visibility; selecting LZs, rally points, pickup points, FARRPs, and other tactical sites; and determining one's own visibility to enemy forces.

Present Deficiencies. Although the importance of masking is clear, no practical methods of accurately determining the masked areas and altitudes from map study have been devised, except for the most obvious situations and solutions. For example, FM 1-1, Terrain Flying, offers only this advice on planning masked routes:

> To do this in mountainous or rolling terrain, plan the route on the friendly side and below the crest of a ridgeline. In very gently rolling terrain, plan the route across the low terrain such as stream beds where it does not serve as an avenue of approach to the enemy position. In arid or open areas, plan the route along stream beds or depressions where trees may exist.

Examination of standard topographic maps indicates that the masking determinations will often be considerably more complex. The procedure for manually plotting masked areas, based on a series of profiles, is described in FM 21-26, Map Reading (16). This procedure entails an extremely time-consuming series of steps to plot the masked areas for even a relatively small geographical expanse. Such an approach is totally impractical for an aviator who needs to determine the masking available in broad and long flight corridors, with several known or suspected enemy positions in the area of operation.

IMPS Capabilities. A computer-generated topographic display may perhaps make its greatest contribution in the computation of masked areas and altitudes for terrain flight. Such computations are relatively simple ones, but the requirement for hundreds or thousands of computations is the arena in which computers are most valuable and efficient. The IMPS system can quickly produce a plot of the areas visible and not visible to an observer or radar at the designated position.

A number of enemy positions could be designated, if desired, to depict the likelihood of being observed given the actual battlefield situation. It remains only for the aviator to choose the most direct path to this objective through the masked areas, and to select the shortest paths between masked areas when brief exposures are unavoidable. Similar computations may be performed to determine radio communication points, fields of fire, and the visibility of checkpoints or tactically important sites.

Figure 3. Examples of masking plots at NOE flight level.

Figure 3 shows a sample portion of a masking plot depicted on the map display. The horizontal bands show the area unmasked from enemy radar. The size of the unmasked areas varies with the altitude of the aircraft--the lower the altitude, the smaller the unmasked areas. Because it is not yet feasible to compute masked areas in real time, the computation of masked areas is done prior to the light, at the mission-planning console.

Although the masked areas must be computed at the mission-planning console in the TOC, the aviator is able to compute intervisibility along a straight line while in flight. The aviator indicates the viewing point and viewed point by entering coordinates or placing a cursor at the desired point on the map display.

The CDU indicates the required altitude above ground level (AGL) at the viewing point for visibility to the viewed point. In addition, it shows whether or not the aircraft would have a terrain "backdrop" when hovering at the intervisibility altitude, and the range between the two positions. "Backdrop" is the case in which the line of sight from a viewed position is terminated by a landform (within the scene memory). The concept of the backdrop is illustrated in Figure 4. The line-of-sight from the tank to the aircraft in this figure is terminated by a landform. Thus, there is a terrain backdrop that greatly increases the difficulty of visually detecting the aircraft. If the aircraft rises to a greater altitude (as shown by the dashed aircraft outline), it would be silhouetted against the sky and would present a relatively easily detectable target to the enemy.

FEATURE SELECTION

Requirements. Because of the many different military activities supported by maps, it is very difficult to make generalizations about the types of features that should be shown on maps. Both the tactical situation and the geographic area determine the importance of various hydrographic, vegetation, and cultural features. If these features could become "key terrain" because of the tactical situation, or are by their nature good navigational checkpoints, they should be portrayed on the map. However, it is not possible to rank hydrographic, vegetation, and cultural features in order of their importance as key terrain or navigational checkpoints without very carefully defining the circumstances. Categories of features valuable in some situations are of little help in others. Although as many potentially useful features as possible should be portrayed, the density of features depicted on the map must not be so great as to create a "clutter" problem, with symbols crowded together, overlapping each other, and obscuring the basic landform contour information.

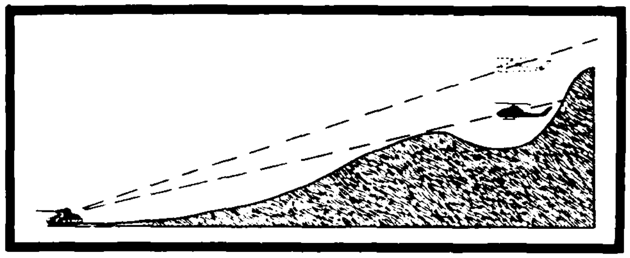

Figure 4. Examples of silhouetting and backdrop.

In order to correlate mission information with the topography in the area of operations, the Army aviator must heavily annotate his map, either directly or with a series of notes and overlays. Dozens of information items are absolutely required, and many more are extremely useful. Once again, it is important that the annotations and overlays do not obscure other important features on the map.

Present Deficiencies. Because of the high cost of producing paper maps, virtually all products of the Defense Mapping Agency are designed to serve the needs of several different classes of users. It would be impossible to produce maps with all information desired by all the potential users without cluttering the maps beyond the point of legibility. Consequently some compromises must be made in each map's information content, so that each class of user is likely to find the map deficient in some manner. Even a map designed specifically for Army aviators could not present all of the potentially useful topographic information because of the clutter problem, and the cartographer is forced to make judgments regarding the items of information best omitted. Paper maps must be oriented north-up to ensure legibility of alphanumerics and other symbols, although other orientations may be superior for navigation and planning purposes.

Aviators have found that direct annotations must be limited in number if the topographic information is not to be obscured, and that a limited amount of annotations are possible with overlays. Many annotations on a single overlay introduces unacceptable clutter, and attempting to use multiple overlays introduces the problems of positioning errors and lost time in overlay selection and alignment. In addition to the clutter problems, the aviators' attempts to copy tactical information from the situation map at the TOC (or from other aviators' maps) introduces two types of error--errors of inaccurate reproduction of the data, and errors of omission of critical items of information because their value is not immediately apparent.

IMPS Capabilities. The IMPS system permits the aviator to select any combination of topographic features and tactical information so that he may design an optimal map display for mission-planning and in-flight use, no matter what type of terrain or battlefield situation he encounters. Various overlays and annotations may be displayed at will or rapidly deleted to study the underlying topographic data. The point symbols created by the IMPS digital map generator are inset in the terrain data so that they are always upright, regardless of map rotation with aircraft heading. This technique provides optimal legibility of alphanumerics and other symbols with any map orientation, as shown in Figure 5. Because the aviator controls the feature selection rules, he is also in control of the density of displayed information and can prevent or eliminate disruptive clutter.

Figure 5. Examples of IMPS point, linear, and area features.

The battlefield situation data produced by the G2/S2 and G3/S3 is entered on the map cassette from a master tape at the TOC, so that the aviator is provided with the most recent data--and the time losses and errors accompanying manual reproducing of situation map data are eliminated. In addition, the problems of overlay selection, positioning, alignment, and smearing are avoided. Furthermore, the aviator may annotate the cassette with planned course lines, checkpoints, and other data, either at the mission-planning console or in the aircraft, and display this information at will.

In summary, the IMPS system permits the aviator to tailor the feature selection rules to best suit his needs depending upon the type of terrain and the level of clutter acceptable on the display screen. Like scale and contour interval selectability, feature selectability would permit the aviator to maximize the accuracy and utility of the terrain portrayal while minimizing the presence of irrelevant information and clutter on the display.

FLIGHT PLAN

Requirements. After mission-planning tasks have been completed, the aviator must annotate his map with data to indicate the selected flight path to and from the objective and, in some cases, alternative routes to be used in the event of changes in the battlefield situation. The aviator normally annotates the map with air control points (ACPs) coincident with easily recognized terrain features. ACPs are often selected by unit commanders, and have the dual role of checkpoints and cues for some tasks, such as turning to a new general heading, communicating by radio, or changing from contour to NOE flight mode. Lines are usually drawn between the ACPs to define the legs of the flight, even though the actual flight path is a weaving one. The length of the legs is measured and the headings are determined by protractor (plotter). The length and heading data may be recorded on the map sheet or in some type of log or kneeboard.

When enroute, the ACPs and course line annotations are used to maintain the appropriate courses and airspeeds for successful performance of the mission. Frequent checks are made to determine the present position of the aircraft with reference to the ACPs, and to determine required changes in aircraft course and speed in order to arrive at ACPs or the objective on time. Timely arrival is particularly critical in crossing friendly lines at scheduled air passage points (to avoid friendly air defense artillery), in following a supporting artillery curtain along the flight route, and in massing for a surprise attack on enemy positions.

Present Deficiencies. Although paper maps are easily annotated, penciled-in lines tend to obscure topographic data, grease pencil annotations often smear, and acetate overlays may be difficult to keep in position. Measurement of the course line lengths and headings is somewhat clumsy, especially when contingency operations require that these tasks be performed at night. Determination of the appropriate headings and airspeeds for timely arrival at ACPs is inexact and inconvenient.

IMPS Capabilities. When it is possible for the aviator to report to the TOC for a pre-mission briefing, he will be able to use the special features of the IMPS ground-based system to perform mission-planning tasks. It is not unusual, however, for the aviator to receive fragmentary orders by radio. For this reason, the airborne IMPS system permits certain annotations to be made in the aircraft, including the entry of ACPs and the planned flight path. In flight, the IMPS system can provide the range and bearing of these ACPs on demand. In addition, the IMPS system can compute the time required to arrive at an ACP at current speed, or the speed required to arrive at a given time.

FLIGHT SIMULATION

Requirements. After performing the route-selection and map-annotation tasks, aviators will use any remaining map-study time to become familiar with the planned route. This familiarization often takes the form of a mission rehearsal and includes attempts to visualize the expected landforms, contemplate planned activities and contingency events enroute, and review the set of responses to these expected and unexpected events.

Present Deficiencies. Mission rehearsal with paper maps presents several problems. Many of these problems, such as the difficulties of contour interpretation, inability to select map scales or feature selection rules, and the near-impossibility of determining cover and concealment, have been discussed in other subsections of this report. Another deficiency in using paper maps is in the difficulty of determining the effects of wind speed and direction on aircraft flight, especially when the aviator must continuously change the heading of the aircraft during NOE flight.

IMPS Capabilities. The capabilities of the IMPS system to interact with the aviator and provide the desired map scales, features, contour visualization aids, and masking plots have been previously described. These capabilities are particularly valuable in the mission rehearsal activities because they assist in route familiarization without the presence of real-world terrain.

The IMPS flight simulation feature also enables the aviator to examine exactly what terrain depictions and areas will be shown on the color CRT during the mission, depending upon his control settings, by presenting a map display moving as if the aircraft were in flight. Movement of the map is based upon the route selected, the aircraft speed, and the wind speed and direction.

An additional benefit of the flight simulation feature is that it permits operation of all of the airborne controls and displays. Thus, part of the mission rehearsal activities may include practice with the IMPS control and display features both to review their characteristics, and to fine-tune their settings for the upcoming mission.

TAPE COPYING

Requirements. Army aviators are expected to move rapidly over long distances in order to surprise the enemy and concentrate forces for decisive combat power. In order to perform their required missions, adequate maps are needed for flight to the new area of operations and additional maps and map overlays are needed to depict the battle area, the friendly and enemy situation, and other battlefield information.

Present Deficiencies. Obtaining the required large-scale paper maps for an area of operations has always been difficult for Army aviators, and the number of map sheets needed for a lengthy flight can cause significant map storage and handling problems in the cockpit. Obtaining and hand-copying suitable map overlays carries a penalty in lost time. Furthermore, the copied overlays are likely to be less accurate with each successive tracing, and critical information may be left out because its importance was not immediately obvious.

IMPS Capabilities. The IMPS system provides a tape-copying capability in each aircraft. The tape-copying feature permits extremely rapid transfer of error-free map and tactical overlay data from one cassette to another. Each cassette will contain map and overlay data for 100 x 100 kilometers--about 16 times the area shown on a standard 1:50,000 scale paper map. Prior to or upon arrival at a new area of operations, an aviator can borrow the required cassette and copy the data on a blank or previously used cassette. This method of dissemination can be used both for communication between superior and subordinate units and for ensuring that all aircraft in a unit have complete terrain and tactical data.

OPERATION OF THE GROUND-BASED SYSTEM

This section of the report describes the procedures employed in operation of the ground-based portion of the IMPS system. All of the functions performed by the airborne system are provided. In addition, the IMPS provides six sets of special functions important for mission planning. The requirements for each set of functions and the present deficiencies in providing these functions are briefly described, followed by short discussions of the capabilities of the IMPS system for meeting the requirements and overcoming the deficiencies. Extensive descriptions are available in a separate report (17).

In the subsequent paragraphs, operational procedures are described for the following special IMPS functions:

Overlay Entry and Editing
Oblique View Construction
Pilot Plan Annotations

OVERLAY ENTRY AND EDITING

Requirements. One of the central activities at a tactical operations center (TOC) is the maintenance of well-edited situation boards and associated special overlays showing such data as enemy situation, friendly situation, fuel and armament locations, hazards and obstacles, battle positions, assembly areas, radar coverage charts, artillery target points, and the scheme of maneuver for operations. Some of these data are from combat information--raw data used for fire and maneuver as received, without interpretation or integration with other data. Another portion of these data are intelligence--data that have been analyzed, validated, or integrated with other data. Much of the information is operations data, either locally generated or received from superior or supported units. Any of these types of information may be critical to successful military operations and must be supplied to aviators in accordance with their requirements for specific mission data.

Present Deficiencies. The primary deficiency in dealing with combat information, intelligence, and operations data is in conveying the critical information to those who need it most--aviators who must perform combat missions. The TOC typically has far more information than the aviator can use for a given mission and selecting just the important information items is a time-consuming task, whether it is performed by TOC personnel or the individual aviators. The situation board overlays are too large and often too cluttered for in-flight use. When portions of these data are copied for specific missions, however, two types of errors are introduced: errors of inaccurate data reproduction, and errors of omission of critical data. In addition, as previously discussed, overlay use presents problems in the aircraft, especially if multiple overlays are required for presentation of all of the useful information.

IMPS Capabilities. The IMPS system permits the storage and selective display of many types of digital overlays. These data bases may be updated rapidly by several types of entry devices. A keyboard, keypad, touch-sensitive screen, or joystick and cursor may be used to enter and erase features in the various data bases. A special tablet for map overlay digitization is also provided. The masking plots are also generated through use of these features.

Any of the data base overlays generated by the IMPS can be selected or deleted in the cockpit by pressing line select keys. Thus, the problems of handling multiple acetate overlays and obscuration of topographic data during flight are eliminated. Furthermore, the IMPS editing capability significantly increases the likelihood that aviators will be provided with the most current, accurate, and complete information available.

OBLIQUE VIEW CONSTRUCTION

Requirements. An oblique view of the terrain is one which portrays the terrain as it might be seen from some angle between ground level and directly overhead. The virtue of oblique photographs is that they present terrain from a more familiar point of regard than that provided by the vertical (plan) view, and features are usually more recognizable. In particular, terrain relief becomes much more discernible in an oblique photograph than in a vertical photograph (except for stereo photographs).

The preflight uses of the oblique view included its employment in route selection, tactical decision-making, and route rehearsal. The oblique view simplifies the specification of good checkpoints, barrier features, battle positions, landing zones, and many other such terrain-related mission-planning requirements. Examination of the features as portrayed by the oblique view offers the aviator an opportunity to develop an "area familiarity" without exposing him to the unacceptable risk of flight hundreds of feet above the actual terrain in a high-threat environment.

Present Deficiencies. Because of the risk and the time requirements involved in obtaining a sufficient number of oblique photographs to adequately cover an area of operations, they are very unlikely to be available to Army aviators.

IMPS Capabilities. A computer-generated topographic display system can, in many respects, substitute for oblique view photography. In many respects, computer-generated visualizations are actually superior to photographic imagery. With the IMPS system, the aviator need not search for or request specific aerial photographs, but can select exactly the views he needs to examine the terrain--either from well above ground level, or from NOE altitude. Such a system may be employed to construct a perspective view of the landforms as they might appear to the human eye, either at ground level or at any chosen elevation above the terrain. Although the current IMPS design does not include cultural, hydrographic, or vegetation features in the oblique views, this capability is being actively explored by AVRADA. An examples of a computer-generated perspective views is shown in Figure 6.

PILOT PLAN ANNOTATIONS

Requirements. Following the operations (G3/S3) and intelligence (G2/S2) briefing, the aviator selects and plots LZs, ambushes and/or firing positions, ACPs, flight routes, potentially hazardous points or areas, key terrain features, and other items of importance to the mission. The special considerations for route selection and annotations have been previously discussed.

The aviator must perform a map reconnaissance, studying the map (and aerial photos, if available) until he is able to visualize the entire route of flight. He must identify areas where detection must be expected because sufficient masking is not available, even at NOE flight levels, and plan for suppresive fires, smoke, chaff, or standoff jamming. The flexibility of mission conduct is in direct proportion to planning efforts, and extensive planning requires detailed map annotations. The ability to mark, draw, and write upon the map provides an unsurpassed aid in recalling the information noted during the planning process.

Present Deficiencies. As noted previously, paper maps are easily annotated, but pencil lines may obscure important topographic data, grease pencil annotations are easily smeared, and acetate overlays may be difficult to keep in position.

IMPS Capabilities. The ground-based IMPS can provide annotation capabilities similar to (and in some respects better than) those currently used on paper maps. The aviator can rapidly insert point symbols of his choosing, "draw" linear and area features of importance to the mission, and insert words when special notations are required. The pilot's annotations are similar, but simpler than those of the editing procedures previously described. In the aircraft, these annotations may be selected or deleted at the touch of a button, so that data obscuration or smearing is no longer a problem.

Figure 6. Slope-shaded terrain perspective view.

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INDICATEUR CARTOGRAPHIQUE POUR HELICOPTERES

par
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0. RESUME

Les hélicoptères de combat modernes sont appelés à évoluer dans un contexte tel que la charge de travail de l'équipage augmente constamment. L'indicateur cartographique électronique est un moyen d'alléger considérablement la tâche de Navigation. Il donne en permanence la position présente, permet des recalages de navigation très rapides. Il peut fonctionner "Nord en haut" ou "Route en haut". Un zoom permet de grossir certains détails. Un joystick sert à déplacer manuellement la carte pour visualiser d'autres zones. Des changements d'échelles sont possibles. Si l'hélicoptère comporte un système de visualisation performant, il est possible de doter l'indicateur cartographique d'une capacité de dialogue permettant de renseigner la carte au sol avant la mission et même pendant le vol.

1. INTRODUCTION

Compte-tenu de l'évolution actuelle, les hélicoptères de combat modernes doivent s'adapter aux situations suivantes :

* le nombre de menaces Sol-Air ou Air-Air est en constante augmentation. L'hélicoptère doit voir sa discrétion améliorée :
  . par sa conception : diminution du bruit, de la signature infrarouge,
  . dans son emploi : en étant capable de vol tactique.
* les missions doivent pouvoir être remplies si possible par tous les temps.
* la mise en oeuvre de systèmes d'armes performants : canon, missiles, roquettes, doit être possible.

Cette évolution se traduit finalement par une augmentation importante de la charge de travail de l'équipage alors que parallèlement la croissance du nombre des menaces nécessite de rendre ce dernier plus disponible pour :

- l'observation,
- la détection,
- l'identification,
- les actions défensives ou offensives.

Un remède consiste à utiliser les ressources offertes par l'évolution technologique pour :

- traiter le maximum de données,
- faciliter et accélérer la prise de décision de l'équipage.

THOMSON-CSF travaille dans ce sens en étudiant et développant des produits tels que :

. visualisations couleur et générateurs de symboles,
. viseur/visuel de casque,
. viseur de tête haute,
. indicateur cartographique.

Ce dernier produit peut fortement alléger le travail de l'équipage. En effet une tâche est particulièrement consommatrice de temps : "la tenue à jour de la Navigation".

Les systèmes de navigation hélicoptères, pour des considérations de prix, ne sont pas des systèmes "haut de gamme" : des recalages de navigation fréquents sont nécessaires, en outre les contraintes du vol tactique et de navigation avec conditions de vision médiocres n'arrangent pas les choses.

La dernière génération d'indicateur cartographique électronique, développée dans ce but, bénéficie de l'expérience acquise dans ce domaine par THOMSON-CSF et des ressources offertes par les plus récentes évolutions technologiques.

Pour une machine équipée de visualisation électronique couleur, cet équipement permet d'avoir :

- la position présente entretenue en permanence (Nord en haut ou Route en haut),
- des recalages du système de navigation simples et rapides,
- des examens de zone de carte avec grossissement,
- des changements d'échelles de carte,
- des examens de zones qui seront survolées ultérieurement.

En option ce matériel peut aussi être utilisé comme une console de visualisation interactive.

Avant le vol, il est possible de mémoriser un certain nombre d'informations concernant la mission :

- délimitation des zones à surveiller,
- report des positions des forces amies connues, ennemies connues ou supposées,
- points de ravitaillement.

En cours de vol, il est possible de noter des informations complémentaires :

- évolution des positions ennemies par exemple.

Les informations sont mémorisées en utilisant les symboles codés OTAN en usage dans les différentes armées pour "renseigner" les cartes papier habituelles.

2. DESCRIPTION DE L'INDICATEUR CARTOGRAPHIQUE

2.1 Rappel (fig 1)

L'indicateur cartographique est un équipement qui, ajouté à un système de visualisation électronique (couleur de préférence), permet de présenter sur un écran de télévision l'image de la carte.

Les coordonnées de la position présente fournies par le système de navigation servent à positionner automatiquement la carte sous un réticule symbolisant la position présente.

Des commandes permettent à l'utilisateur de :

. sélectionner l'échelle,

. sélectionner le mode : - Nord en haut,
  - Route en haut.

. zoomer la carte,

. déplacer un réticule de désignation (joystick),

. valider un recalage de navigation,

. visionner une zone différente de celle survolée,

. en option utiliser le système comme une console interactive.

La carte est stockée dans une cassette amovible insérée manuellement par l'utilisateur.

THOMSON-CSF a développé successivement plusieurs générations d'indicateurs cartographiques. La dernière mise au point pour le Programme Français d'avion de combat MIRAGE 2000 comporte un dispositif d'analyse de carte entièrement électronique. L'utilisateur bénéficie d'un équipement très performant et fiable.

Une version dérivée a été adaptée aux besoins spécifiques "Hélicoptères".

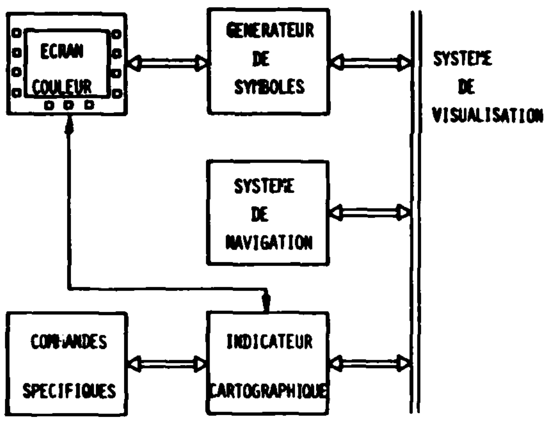


FIG 1

Dans les systèmes simples, l'indicateur cartographique peut comporter une capacité propre de génération de symbologie, évitant ainsi l'emploi d'un générateur de symboles.

2.2 Stockage de la carte

Le premier problème à traiter est le stockage de la carte. L'évolution des capacités de stockage des dispositifs à mémoire numérique permet d'espérer que dans l'avenir il sera possible de stocker des cartes "digitalisées" mais pour les années qui viennent le film couleur reste de très loin la "mémoire" la plus économique.

La figure 2 montre le principe retenu pour la réalisation d'un film couleur de 35 mm d'une longueur d'environ 20 m.

La (ou les) carte Papier concernée est dans une première étape photographiée en couleur sous forme de "n" microcartes (avec de légers recouvrements).

Le film est réalisé dans une deuxième étape à partir des microcartes en photographiant vue par vue des zones telles que "A" fig 2. Chaque zone assure un recouvrement des zones voisines (55 %).

Les zones sont photographiées successivement pour constituer sur le film des bandes de terrain telles que 1, 2, 3.

Lorsque le film de la région concernée à l'échelle est réalisé, on répète le même processus en partant de la carte de la même région à l'échelle Y. Les vues correspondantes sont placées à la suite des premières (processus automatisé, contrôlé par microprocesseur).

Le film terminé est logé dans une cassette (fig 3). Une mémoire programmable, solidaire de la cassette, contient les données d'identification de la région photographiée.

Lorsque l'utilisateur insère la cassette dans l'indicateur cartographique, la lecture de cette mémoire permet l'initialisation immédiate du système cartographique.

2.3 Fonctionnement de l'analyseur de film

L'équipement "indicateur cartographique" sert à restituer une image couleur extraite d'un film.

La figure 4 aide à comprendre le principe de cette restitution.

Les coordonnées de la position présente (fournies par le système de navigation) permettent à l'indicateur cartographique de dérouler (servo-mécanisme) le film jusqu'à ce que la position présente se trouve dans le cadre C fig 4 du dispositif d'analyse.

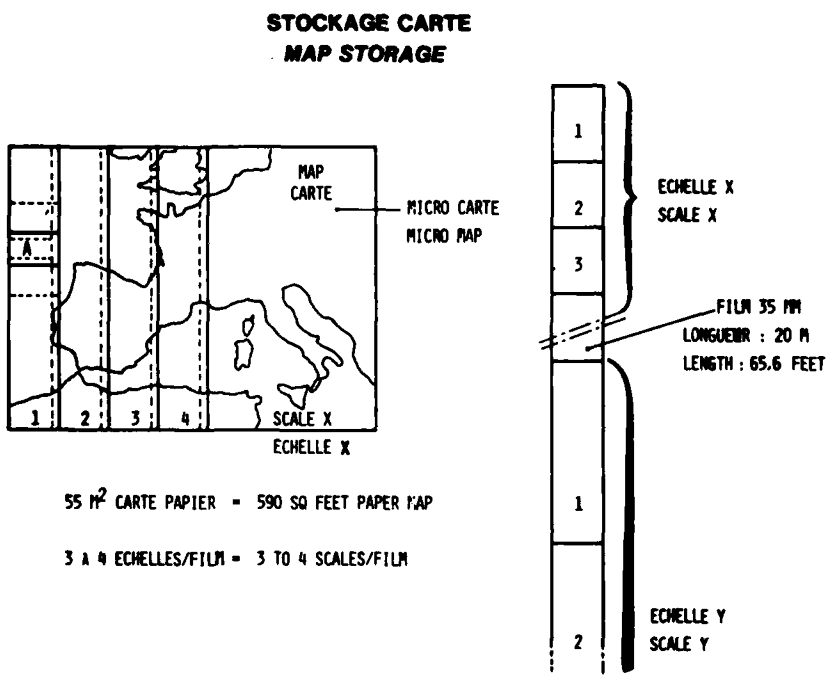


FIG 2

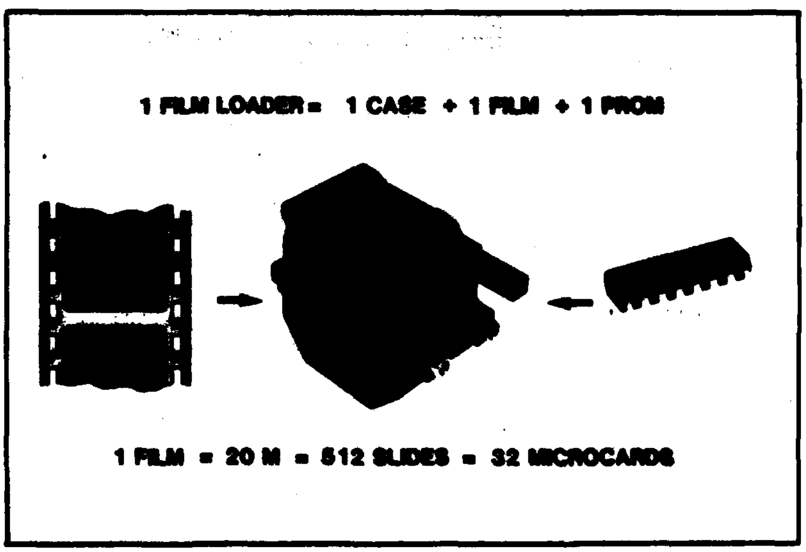


FIG 3

Deux exemples sont présentés sur la figure 4 :

Exemple 1 : la position présente est symbolisée par la croix centrale de la fenêtre d'analyse délimitée par la zone claire. A la prise de vue, le film est réalisé avec les cartes usuelles "Nord en haut". La fenêtre d'analyse est orientée comme le film : Nord en haut.

Elle représente sur le film la zone qui va être balayée très rapidement par un spot électronique grâce à une technique dite de "Flying spot". Ce spot est généré par un tube cathodique en lumière verte à luminosité constante.

Une fois la fenêtre balayée horizontalement par le spot en "N" lignes selon la même méthode que celle employée pour un téléviseur domestique, le balayage recommence en haut et à gauche de la fenêtre.

Cette dernière (et non le film) se décale latéralement de la même façon que la position présente en agissant sur les commandes électroniques de balayage (longitudinalement le film se déplace). L'exemple 1 illustre une application sans utilisation du zoom = rapport 1 - Nord en haut.

L'exemple 2 illustre une application où l'utilisateur a sélectionné :

. le mode route en haut, de ce fait la fenêtre est orientée dans le sens de la route suivie (toujours en agissant sur les commandes électroniques de balayage),

. un zoom de rapport 2,5 environ : ceci a pour effet en agissant sur les commandes électroniques de balayage de réduire les dimensions de la fenêtre.

L'utilisateur perçoit sur son écran une image qui occupe toujours toute la surface de l'écran grâce à un "grossissement" électronique.

Cette figure 4 permet d'illustrer les points suivants :

- le passage du mode Nord en haut au mode Route en haut est totalement électronique,
- l'effet de zoom est entièrement électronique.

Ces points fondamentaux illustrent les progrès considérables réalisés.

L'utilisateur est le principal bénéficiaire de la souplesse d'emploi et de l'amélioration de fiabilité obtenues par rapport aux générations antérieures où ceci était réalisé mécaniquement.

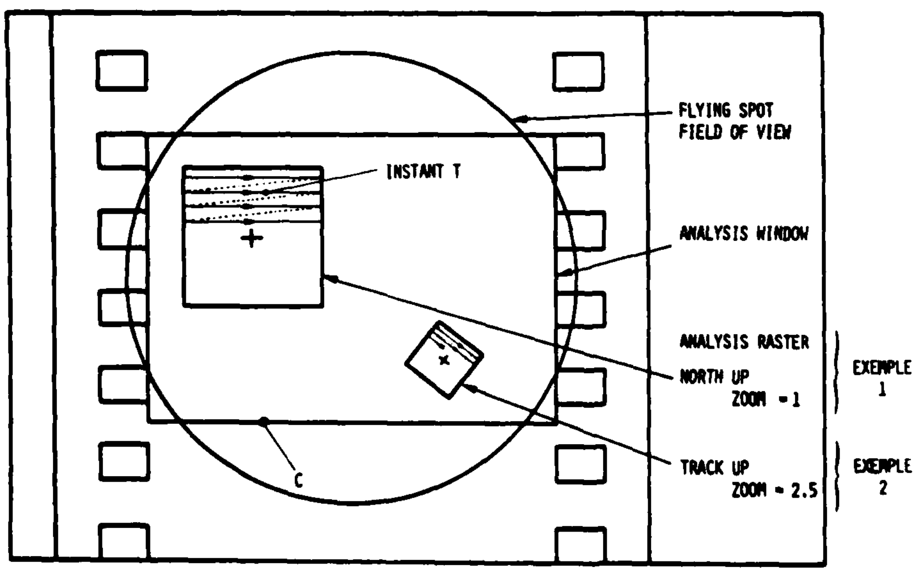


FIG 4

Le principe de l'indicateur cartographique électronique ou Analyseur de Film est présenté fig 5.

Les coordonnées de la position présente fournies par le système de navigation sont reçues et traitées par INT (interface) communiquées au microprocesseur qui commande le servo-mécanisme de positionnement de film (SERVOING).

La position présente sur le film est centrée sur la fenêtre d'analyse.

Le microprocesseur contrôle les commandes de balayage du tube cathodique du Flying spot.

A chaque instant le spot issu du Flying spot (lumière verte - luminosité constante) est focalisé sur le film couleur par un objectif (OBJ). Ce spot balaye la trame d'analyse dont la dimension peut être modifiée pour l'effet zoom.

Après avoir traversé le film, le spot (modulé par les informations mémorisées sur le film couleur) traverse un condenseur (COND) puis un séparateur couleur (Rouge - Vert - Bleu). Les trois faisceaux rouge - vert - bleu résultants sont amplifiés grâce à trois photo-multiplicateurs (PM) ; à la sortie d'un module de traitement (SIGNAL PROCESSEUR) on obtient le signal Vidéo qui permet d'obtenir une image TV sur le système de visualisation de la planche de bord.

Le diagramme synoptique de l'indicateur cartographique Mercator est présenté fig 6.

A noter qu'une mémoire d'image peut être utilisée en option lorsqu'il faut :

- superposer à la carte une image radar par exemple,
- "geler" une image pendant le temps de recherche d'une autre,
- présentation d'images différentes pilote-copilote.

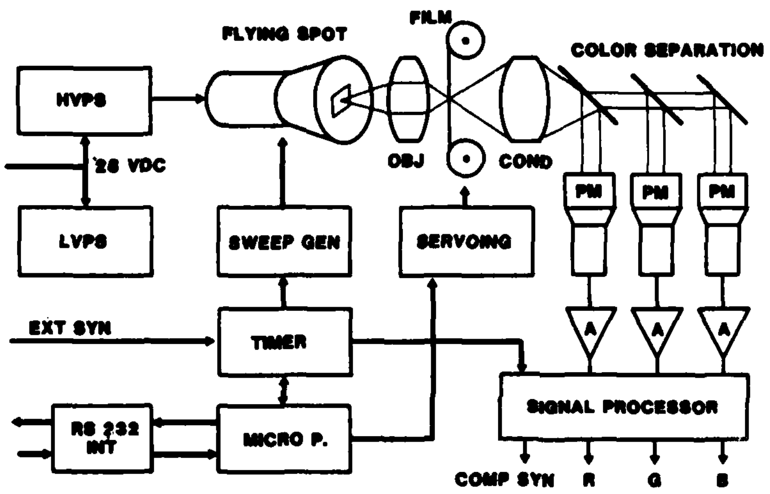


FIG 5

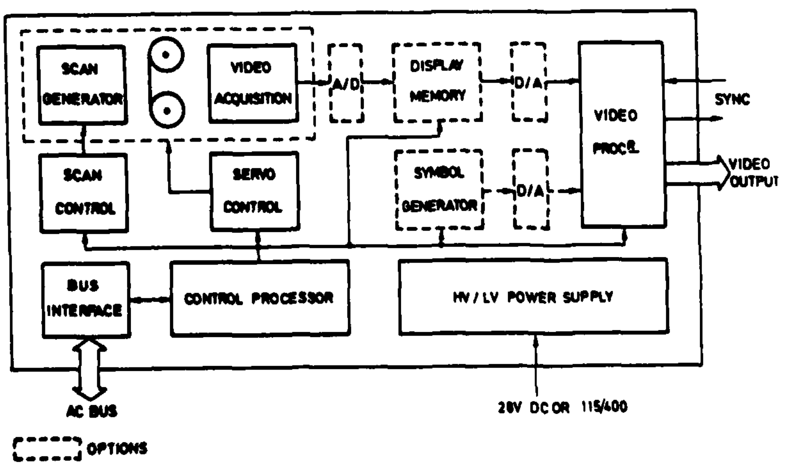


FIG 6

2.4 Caractéristiques

Les caractéristiques ci-après concernent le matériel développé en France. Un accord de license a été signé avec la firme américaine Hamilton Standard qui conduira à un produit répondant aux normes U.S.

2.4.1 Caractéristiques mécaniques - fig 7

- format : 1/2 ATR 380 (12,4 x 19,3 x 38 cm)
- Poids : 12 kg
- Refroidissement : air forcé.

2.4.2 Alimentation électrique

- 28 V DC/150 W MIL Std 704 A.

2.4.3 Interfaces de sortie

a/ sortie vidéo - à choisir entre :
- EIA RS 330 (525 1/60 Hz entrelacé 2/1)
- CCIR (625 1/50 Hz entrelacé 2/1).

b/ vidéo monochrome avec option polarité positive ou négative.

c/ synchronisation extérieure possible.

d/ symbologie synthétique TV : incrustation TV possible sur 512 x 512 pts avec 7 niveaux de couleur.

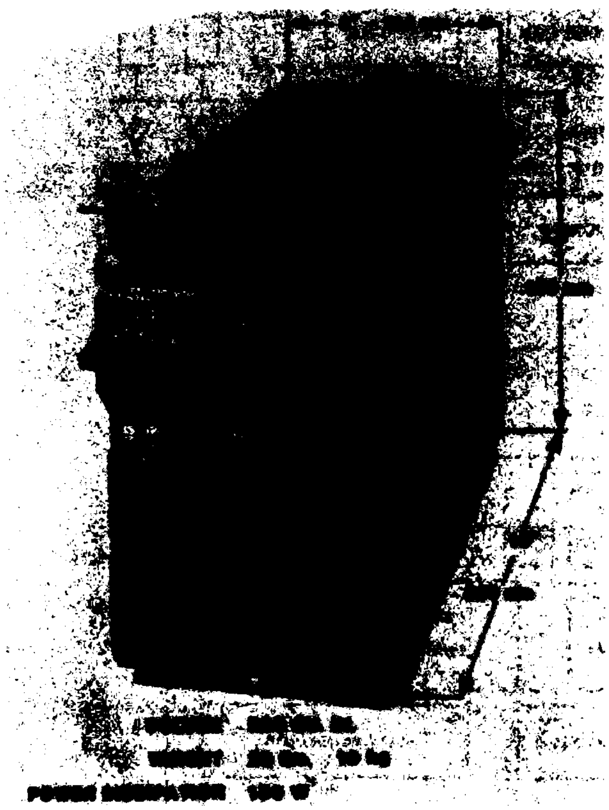


FIG 7

2.4.4 Interfaces d'entrée

Selon besoin, choix entre :

- BUS 1553 B redondant,
- Interface standard RS 232 C.

2.4.5 Fonctions cartographiques

- zoom électronique rapport 1 à 2,5
- rotation d'image pour mode Nord/Route en haut
- position présente centrée ou décentrée
- jusqu'à 4 échelles exploitables.

2.4.6 Autotest

La fonction autotest permanent est capable de vérifier 95 % des circuits internes. Un autodiagnostic est présenté sur la visualisation par le générateur de symboles incorporé lorsque la procédure de test est appelée.

2.4.7 Performances dynamiques

Le film est positionné à 10 microns.

L'erreur de position entre la carte et la symbologie correspond, à titre indicatif à environ 1 mm max sur l'écran, avec σ = 0,3.

Déroulement du film de bout en bout $\leqslant$ 12 s.

3. EMPLOI OPERATIONNEL

3.1 Réalisation des films

Les films sont réalisables à l'échelon d'un pays utilisateur à partir d'une installation qui peut être proposée et exécutée ou simplement spécifiée par THOMSON-CSF. La surface de territoire couverte par une cassette est considérable, bien entendu elle dépend des échelles et de leur nombre/cassette.

A titre d'exemple un film avec l'échelle :

- 1/1000.000 couvre un territoire de 7 400 km x 7 400 km
- 1/ 250.000 " " " de 1 850 km x 1 850 km
- 1/ 100.000 " " " de 700 km x 700 km.

3.2 Version de base

L'emploi de la version de base est simple. Il se traduit pour l'utilisateur par :

- l'introduction ou le retrait de la cassette film - voir fig 7,
- l'utilisation du poste de commande qui comporte :
  . un joystick pour déplacer un réticule de désignation ou déplacer la carte vers une autre zone,
  . une commande de mode Nord/Route en haut,
  . une commande de changement de l'échelle,
  . une commande de validation de recalage de NAV.

3.3 Version optionnelle interactive

Moyennant le choix de certaines options (à préciser avec l'utilisateur en fonction des besoins) il est possible d'utiliser le système indicateur cartographique avec :

a/ renseignement de la carte avant la mission et debriefing,

b/ renseignement de la carte pendant la mission.

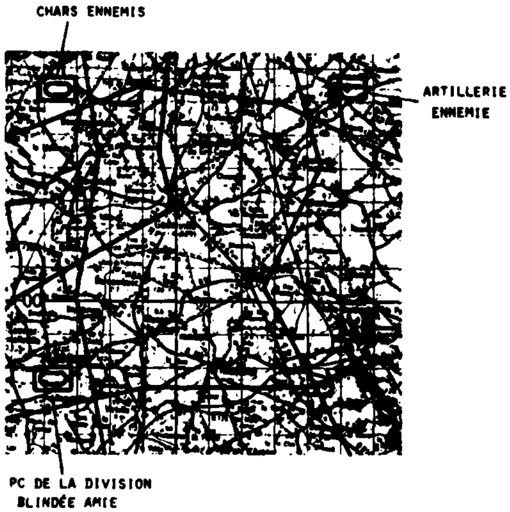
La figure 8 illustre un exemple de ces possibilités qui peuvent permettre :

- de tracer des lignes brisées pour séparer des zones de terrain,
- faire apparaître des symboles répertoriés (ami = bleu ; ennemi = rouge) destinés à positionner des forces identifiées avant ou en cours de mission : des points de ravitaillement, des buts, etc ...

La symbologie est liée à des coordonnées géographiques, elle se transpose en fonction de l'échelle.

Si l'on veut pouvoir renseigner la carte avant la mission, il faut prévoir :

- un module d'insertion de paramètres (mémoire amovible - effaçable),
- une console de préparation spécifique permettant de :
  . visualiser la symbologie que l'on veut enregistrer dans le module d'insertion des paramètres,
  . exploiter au sol les informations enregistrées au cours du vol.



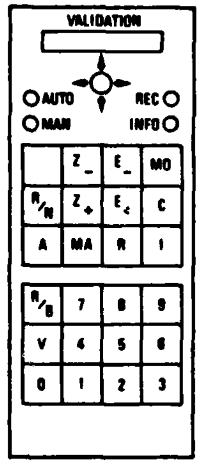


FIG 8

A bord de l'hélicoptère, l'opérateur doit disposer d'un pupitre mobile d'accès au système - voir exemple fig 8 - constitué :

- d'un joystick et d'une commande de validation,
- d'un clavier de commandes de fonctions :
  . zoom / Echelles / Nord / Route / Auto / Manuel / Recalage / Tracé.
- d'un clavier d'accès à des symboles mis en mémoire avec choix Rouge/Bleu.

Cet exemple illustre quelques possibilités de ce système qui doit être défini en fonction des besoins de l'utilisateur.

MAP DISPLAY FOR HELICOPTERS

by
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0. ABSTRACT

Modern combat helicopters have to be adapted to a context of constantly increasing crew work load. The electronic map display is a way to greatly reduce Navigation tasks. It continuously provides present position of the aircraft and allows for very fast navigation up-dating. North-up and Track-up modes are also possible. A continuous zoom is used to enlarge some details. A joystick is used to manually move the map for visualization of other zones. Different maps scales are available. If the helicopter's display system is powerful enough it is also possible to create an interactive console giving the crew the ability to memorize data on the map before the mission and in real time during the flight.

1. FOREWORD

Taking into consideration the evolution of the battle field, modern combat helicopters have to be adapted to the following new situations :

* increasing number of Ground to Air or Air to Air threats.
  The helicopter's discretion must be improved :
  . in its design : reduction of noise and infrared signatures
  . in its use : in being able to fly nap of the earth,
* missions must be performed in all weather,
* the use of performing weapons systems has to be possible : gun, missiles, rockets.

This results in a large increase in crew work load. At the same time the crew has to be more available for :

- observation,
- detection,
- identification,
- defensive or offensive actions.

One solution consists in using the resources of technological evolution :

- to process the maximum amount of data,
- to ease and accelerate the crew's decision making process.

THOMSON-CSF works in those areas by developing equipment such as :

. color displays and symbol generators,
. helmet sight and display systems,
. head-up displays,
. electronic map readers.

Since navigation tasks are particularly time consuming. This last equipment can greatly reduce crew work load.

Navigation systems for helicopters, if low-cost, cannot be too sophisticated. Frequent up-datings are required, and nap of the earth flight in poor visibility do not facilitate the crew's mission.

The last generation of electronic map readers, developed in this context, with the experience acquired by THOMSON-CSF in that field, uses all of the most recent technological advances.

For a helicopter with an electronic color display system this equipment provides :

- permanent present position (North-up or Track-up mode),
- rapid and simple navigation up-dating,
- zone enlargement capability,
- scale change availability,
- possibility to visualize other zones.

Optionally this equipment can be used as an interactive console. Before the flight, it is possible to memorize data concerning the mission :

- definition of zones to patrol,
- pointing of known friendly positions, known or alledged enemy positions,
- maintenance or refueling positions ...

For example, during the flight it is possible to memorize the advancement of enemy forces.

Data is memorized using the standard NATO symbology used for artillery, tanks... ect.

2. MAP DISPLAY DESCRIPTION

2.1 See (fig 1)

The map reader provides an electronic display system (preferably in color) the capability to display a map on a television screen.

Coordinates of present position given by the navigation system are use[illegible] on the map correctly under a fixed reticle.

The crew uses controls to :

- select the scale,
- define the mode : North-up or Track-up,
- zoom the map,
- move a designation reticle (joystick),
- enter a navigation up-date point,
- oversee another zone,
- optionnally use the system as an interactive console.

The map is stored in a removable cassette manually inserted by the user.

THOMSON-CSF has successfully developed several generations of map readers. The last one, used for the MIRAGE 2000 combat aircraft consists of a fully electronic remote map reader. The user has the combined benefits of high performance and reliability.

A derived version has been adapted to the specific needs of helicopters.

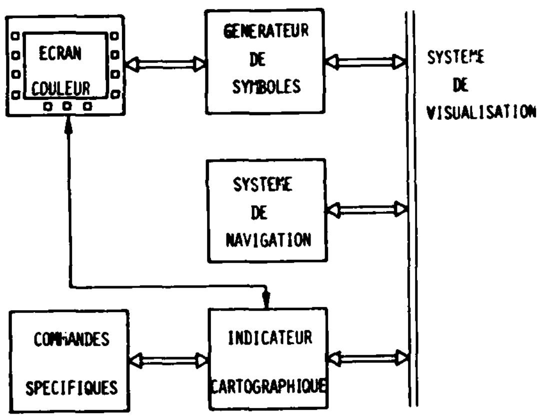


Fig 1

In simple systems, it is possible to give the map reader its own symbol generator capability to avoid using a separate one.

2.2 Map storage

The first step for such a system is the map storage. In the future it will probably be possible to store digitalized maps in large numerical memories, but for the coming years the color film stands as the cheapest and most compact memory base.

Fig 2 shows the principle of a 35 mm color film system. Film lenght is about 70 feet.

In the first step the color map is photographed in "n" micro-maps (with some overlap).

In the second step the film is created from micro-maps, frame by frame as pictured in "A" fig 2.

Each zone provides overlaps of about 55 %.

Zones are photographed successively on film strips like 1, 2, 3 ...

As soon as a given area at the proper scale is completed, the same process is repeated at another scale Y (if necessary). Corresponding views follow on the film (automated micro-computer process).

Once finished, the film is inserted in a cassette (fig 3).

A programmable index memory fixed on the cassette provides the system with data to identify the stored area.

When the user inserts the cassette into the equipment, the system is ready for operation after reading the index memory.

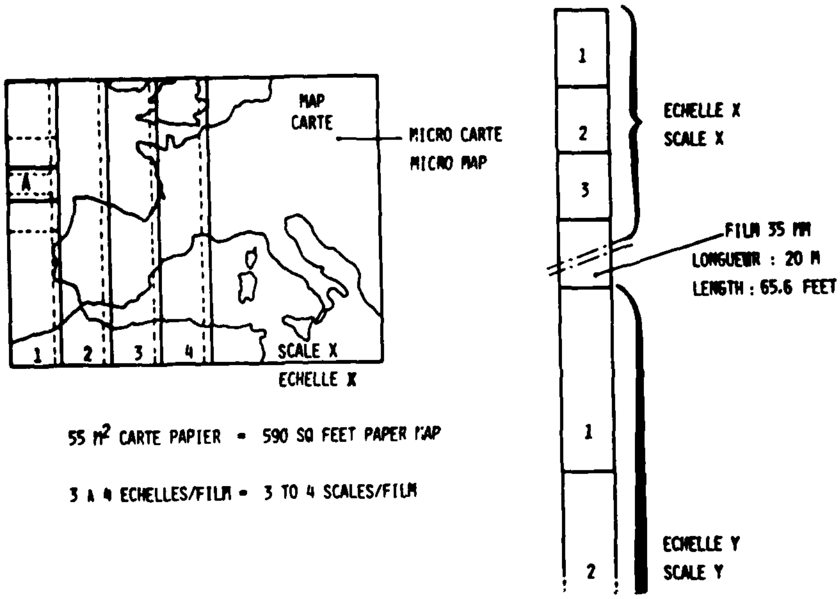


fig 2

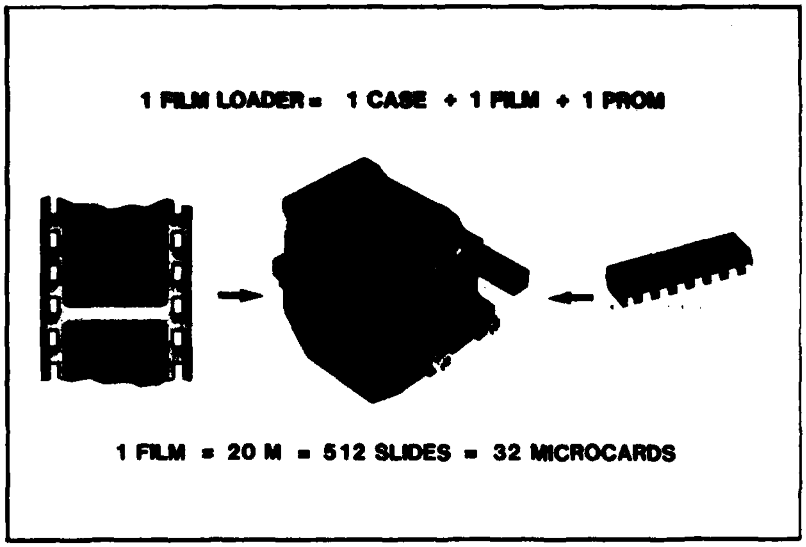


fig 3

2.3 Map reader operation

The map reader equipment is used to retrieve a color raster image from a color film. Fig 4 is helpful for understanding.

Coordinates of the present aircraft position (given by the Navigation system) are used to position the film (through a servo-mecanism) until the proper frame is in the window C. See fig 4 (analysis window).

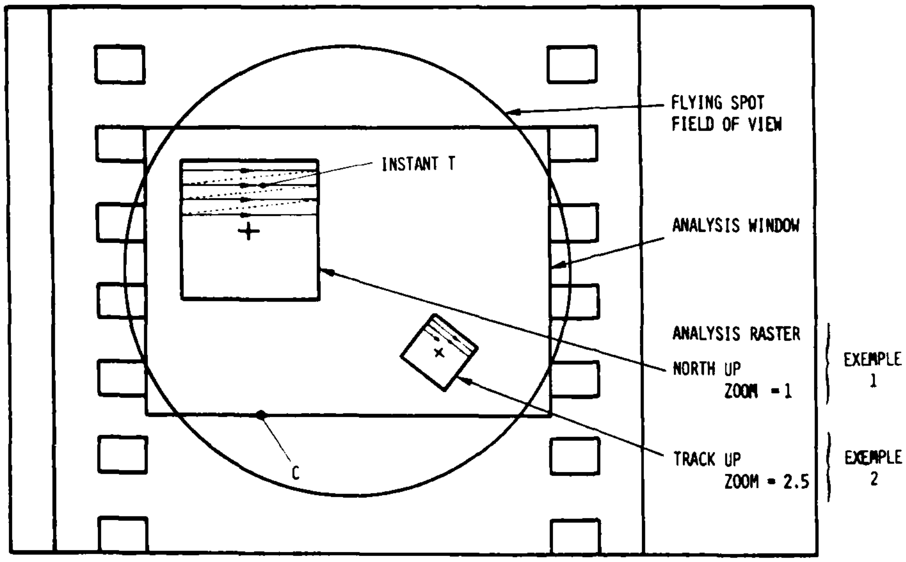


fig 4

Two examples are presented on fig 4.

Example 1 : the present position is symbolized by the cross in the center of the square window. When created the film is made from North-up maps. The square analysis window is oriented like the film : North-up.

The square window represents, on the film, a zone which is rapidly scanned by an electronic spot by "flying spot technique". A cathode ray tube working in green light with constant luminosity generates this spot.

Once the window has been horizontally analyzed in "N" lines (like in a commercial TV set), the electronic scan starts again left-up of the window. The raster scan (and not the film) moves laterally to the new aircraft position due to the electronic sweeping controls (longitudinally,it is the film that moves). Example is is equivalent to an application with zoom/ratio = 1 and North-up mode.

Example 2 shows an application where the user has selected :

- Track-up mode, analysis window is oriented Track-up,
- a zoom/ratio of 2.5 : reduced size of analysis window.

Nevertheless the user always has the same size image on the screen due to electronic enlargement.

Fig 4 shows the following points :

- the change from North-up to Track-up mode is fully electronic,
- the zoom effect is also fully electronic.

These fundamental points illustrate the major advancements which provides the user with great improvements in reliability and ease of operation in comparison with the old mechanical systems.

The map reader principle is presented on fig 5.

Coordinates of present position given by the Navigation system are received by INT (interface) and transfered to the micro-computer which controls the servo-mecanism for film positioning (SERVOING).

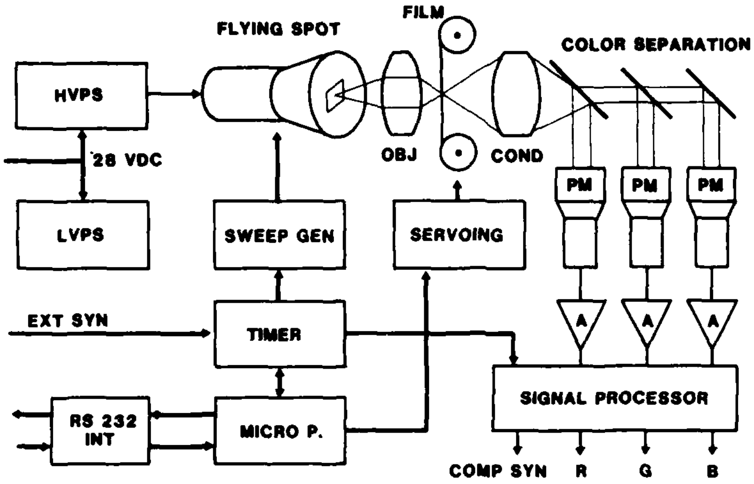


fig 5

The present position on the film is the center of window analysis.

THe computer controls the scanning of the flying spot.

At any time the spot issued by the flying spot CRT (green light - constant luminosity) is focused on the color film by a lens (OBJ). This spot sweeps the analysis window whose size can be modified for zooming effect.

Once it gets through the film, the spot (modulated by color information recorded on the color film) goes through a condenser (COND) and a color separator (Red - Green - Blue).

The three resulting beams are amplified by three photo-multipliers (PM). A video signal is obtained as an output from a processing module (SIGNAL PROCESSOR) and used to produce the TV image on the display system of the instrument panel.

The block diagram of Mercator Map reader is shown in fig 6.

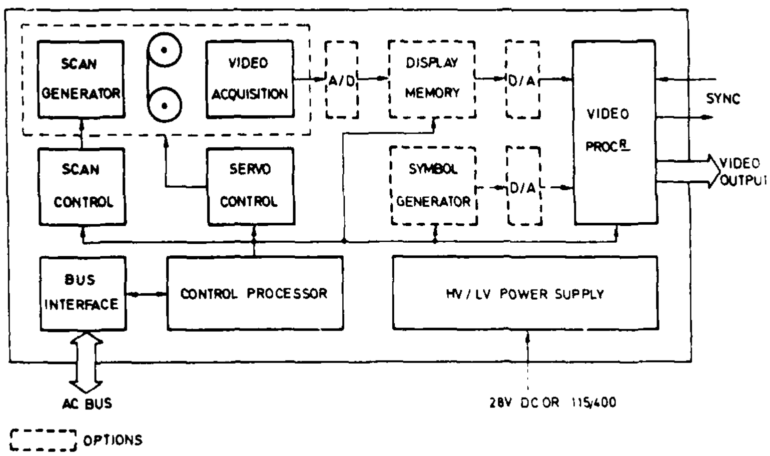


fig 6

Optionally an image memory can be used :

- to superimpose a radar image,
- to "freeze" an image when looking for another one,
- to present different images to pilot and copilot.

2.4 Characteristics

The following characteristics concern a French built map reader. THOMSON-CSF has also signed a license agreement with HAMILTON Standard (UNITED TECHNOLOGIES) to develop a map reader to US standards.

2.4.1 Mechanical characteristics (fig 7)

- size : 1/2 ATR 380 (12.4 x 19.3 x 38 cm)
- weight : 12 kg
- cooling : forced air.

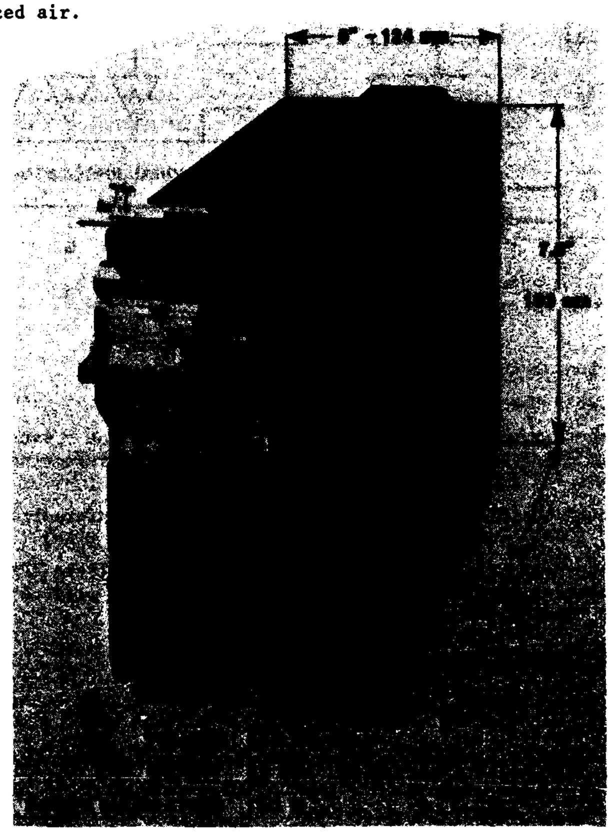


fig 7

2.4.2 Power supply

- 28 V DC/150 W MIL Std 704 A.

2.4.3 Output interface

a/ Video output :

- EIA RS 330 (525 1/60 Hz 2/1 interlaced) or
- CCIR (625 1/50 Hz 2/1 interlaced).

b/ monochromatic video with option between positive or negative polarity,

c/ external sync possible,

d/ TV synthetic symbology : possible on 512 x 512 pixels with 7 color levels.

2.4.4 Input interface

- BUS 1553 B redundant or
- Standard interface RS 232 C.

2.4.5 Cartography

- electronic zoom ratio 1 to 2.5
- image rotation for North-up to Track-up mode,
- present position centered or decentered,
- up to 4 scales available.

2.4.6 Autotest

- Permanent autotest is available to verify 95 % of the internal circuitry,
- Diagnosis is presented on the display by the incorporated symbol generator when the initiated test is selected.

2.4.7 Dynamic performance

- film positioning accuracy 10 microns,
- error positioning between map and symbology could be about 1 mm max on the screen with $\sigma = 0.3$
- end to end film positioning in about 12 s.

3. OPERATIONAL USE

3.1 Film generation

Films are produced in the user's country with an installation which can be proposed and executed or simply specified by THOMSON-CSF. The ground surface stored in a cassette is considerable, of course it is dependant on scales and the number of them in a cassette.

As an example, a film with a scale of :

- 1/1,000,000 covers 4,600 x 4,600 (miles)
- 1/ 250,000 " 1,150 x 1,150 (miles)
- 1/ 100,000 " 430 x 430 (miles)

3.2 Basic version

The use of the Basic version is very simple. For the user, it requires :

- introducing or removing the film cassette (fig 7),
- using a control panel consisting of :
  - . a joystick to move the designation reticle or the map,
  - . a mode control for North-up or Track-up,
  - . a control for scale change,
  - . a validation control for NAV up-dating.

3.3 Interactive optional version

Depending on the choice of options (to be defined with the user, taking into account his operational requirements) it is possible to use the map display system to :

a/ up-date the map before the mission and briefing,

b/ up-date the map during the flight.

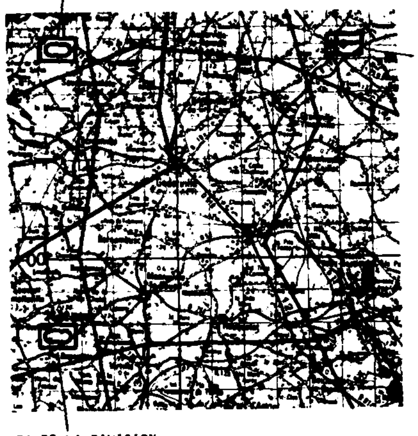
Fig 8 gives an example of these possibilities used for :

- drawing lines to define scouting zones on the ground,
- calling on special NATO symbols (friend = blue / foe = red) to plot identified forces before or during the mission, targets, refueling sites ... ect.

The symbology is linked to the geographic coordinates, it is recorded at different scales.

Up-dating of the map before the mission requires :

- a removable module to insert parameters (removable and erasable memory),
- a dedicated console for :
  - . visualization of the parameters to be recorded in the removable memory module,
  - . visualization on the ground, after the flight, of data recorded during the mission.



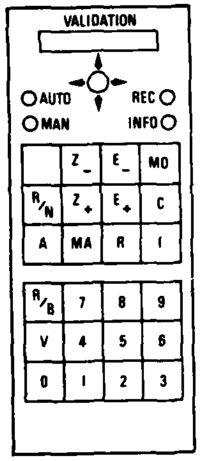


fig 8

During the flight the copilot uses a mobile control box (see example fig 8) made of :

. a joystick and a validation command,

. a function key board for :

- zoom / scales / North-up / Track-up / Automatic / Manual up-dating / Drawing.

. a symbol key board for NATO symbols with the red or blue color choice.

This example gives some ideas of the system capabilities which has to be defined in detail with the user.

-:-:-:-:-:-:-:-:-:-:-:-:-:-:-:-:-:-:-:-

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DEVELOPMENT OF A HELICOPTER INTEGRATED NAVIGATION SYSTEM

by

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SUMMARY

The Canadian Department of National Defence has initiated a project to develop and test a helicopter integrated navigation system (HINS) that is capable of satisfying all the operational requirements of shipborne, anti-submarine warfare helicopters. The HINS mission requirements, development plan and its basic hardware and software configurations are discussed followed by a review of preliminary performance results obtained to date.

The integrated system is expected to bring forth vital benefits in mission reliability, operational efficiency and navigation accuracy. Additional benefits are in terms of the size, weight, height and cost, etc. The integrated system shall possess a high degree of flexibility, so that it can be readily reconfigured to respond to future evolution in Canadian Forces operational requirements and also to track advances in navigation technology.

In the current phase of this work, extensive simulation software packages have been generated to accurately represent the operational characteristics of various off-the-shelf candidate subsystems. An Integrated System Evaluation Program has been developed which takes into account the various characteristics of the navigation subsystems and the mission requirements to systematically evaluate a sequence of candidate configurations. Complex software routines are being developed for the HINS processor to perform the tasks of optimal sensor integration, navigation, operator interface control, data display and system diagnostic self-test.

The examination of candidate configuration performance results and subsequent tradeoff study will enable the designer to determine a preferred HINS configuration which can most effectively satisfy the operational requirements of the marine warfare helicopter.

1.0 INTRODUCTION

The roles of the military maritime helicopter include search and rescue, ocean surveillance, anti-submarine warfare (ASW), and weapon delivery. Many of the missions must be carried out at ultra-low altitudes under all weather and visibility conditions. The increased range, speed and accuracy of modern weapon systems impose stringent accuracy and reliability requirements upon the aircraft navigation system. To enhance mission success in a hostile environment, the pilot amongst other things needs to operate weapon systems, target acquisition and designation systems, radar detection, night vision systems and perhaps engage in air-to-air combat. The traditional manual dead reckoning tasks can no longer provide the required performance accuracy and would unnecessarily distract the pilot from performing mission-critical functions.

More specifically, the Canadian Navy's Sea King helicopter will be nearing the end of its useful life by the beginning of the next decade. As a result the Canadian Department of National Defence (DND) has begun studying options, including the update or replacement of the maritime helicopter fleet. A number of research and development projects have been initiated to develop certain avionic systems. One of these projects is to develop and test an integrated navigation system that is capable of satisfying the helicopter mission requirements within the cost limitations of the program. This project is called the Helicopter Integrated Navigation System (HINS). This paper describes the development plans and the basic software and hardware configurations of the systems under study. Some preliminary results of performance simulation analyses are also presented.

2.0 MISSION REQUIREMENTS

In the 1990's and beyond, the effort in Anti-Submarine Warfare (ASW) will be directed toward extending the range of detection and prosecution of submarine contacts. This may include ships towing sonar arrays and ASW helicopters working in concert with the ships. The steady improvement in the submarine's capabilities means that the CH-124 Sea King, which is currently deployed from Canadian ships,

will eventually lack the range, endurance, detection and data-processing equipment necessary to localize and prosecute the long-range submarine contacts. The two prime solutions for this problem are to update and life-extend the Sea King or replace it with a more modern maritime helicopter. In either case, there exists a requirement for a number of new avionic systems, one of which is a new navigation system.

For the ASW mission the helicopter navigation system must maintain stable and accurate tactical plots over long periods of time. In the anti-surface ship targetting role high orders of absolute and relative navigational accuracy are vital to rapid and successful action. There are further complicating factors as well. Operations must often take place under radio silence and shore-based or satellite navigation aids may be destroyed or jammed during wartime. The small crew of the helicopter must not be burdened with monitoring the functioning of, or updating, the navigation system. Consideration of these factors has led to the following minimum operational accuracy requirements:

I. Radial Position Error (95%):

- with external aids*	2.0 nautical miles (nm)**
- without external aids	1.5 nm/hr

II. Radial Velocity Error (95%):

- with external aids*	3.0 ft/sec**
- without external aids	4.0 ft/sec

III. Attitude Error (95%): 0.5 deg

IV. Heading Error (95%): 0.5 deg

* External aids are those systems such as Omega, Loran and the Global Positioning System (GPS) which rely upon transmitters which are located external to the aircraft and may be unavailable during wartime. INS, Doppler and Radar are representative of internal or self-contained aids.

** It is recognized that these performance levels can be exceeded by a large margin if, as expected, GPS is part of the HINS.

3.0 HINS DEVELOPMENT PLANS

Modern day avionics systems are becoming increasingly complex as the demands for better performance and higher reliability continue to escalate. These demands, however, are being pressed in an extremely cost-conscious environment. The HINS development project addresses the development of the helicopter integrated navigation system by both functionally and operationally integrating the navigation subsystems. This integration is the key to satisfying the HINS performance and reliability objectives in the most cost-effective manner.

With several navigation subsystems available for HINS, a large number of equipment configurations are possible. The typical approach is to use previous experience in selecting two or three candidate configurations in an ad hoc manner. This has the potential danger of eliminating good alternatives early in the project and could eventually result in a suboptimal configuration. Thus, DND has decided to spend a significant portion of the navigation system development time to simulate and study a number of potential configurations with the aim of identifying, developing and testing an integrated navigation system which best satisfies the requirements established for the project.

The HINS approach to achieving this aim is to first perform preliminary analysis and simulation to identify four or five candidate configurations that meet the mission requirements. From the detailed performance analysis of these configurations one of them will be selected for advanced development. The product of this advanced development will be thoroughly tested. The completed navigation system will then be ready for incorporation in the maritime helicopter.

To meet the schedule requirements of the helicopter program the HINS project was divided into two phases as follows:

3.1 Phase 1 System Definition and Design (Figure 1)

1) Survey existing navigation sensors (off-the-shelf and those under development that will be available in the next 5 years) and determine suitability for helicopter integrated navigation systems.

2) Identify a cost-effective set of HINS sensor candidate configurations that can satisfy the performance and reliability requirements of the maritime helicopter.

3) Generate sensor error models, integration (Kalman filter) algorithms and simulation software to assess the HINS candidate configurations.

4) Conduct performance assessments of the candidate configurations using covariance and Monte Carlo analysis techniques.

5) Identify the preferred configuration with which to proceed to advanced development.

As part of the study, extensive work was directed toward collection of data describing potential navigation equipment for the maritime warfare helicopter. This data base was used in an Integrated System Evaluation Program (ISEP) to generate a large number of navigation system sensor configurations. From these configurations four have been selected as candidate system configurations for more complete evaluation during the remainder of the study.

Complex software routines are being developed for the HINS processor to perform the tasks of optimal sensor integration, navigation, operator interface control, data display, and system diagnostic self-test. These will provide DND with the detailed design of an integrated navigation system which can most effectively satisfy the mission requirements of a maritime warfare helicopter.

3.2 Phase II Development and Testing

1) Acquire the sensor hardware for the preferred configuration and complete the development of the design produced in Phase I.

2) Develop the data bus and interface electronics to connect the sensors.

3) Refine the preliminary forms of the Kalman filter and integration software from Phase I into flyable programs. Develop the control and display software and transfer these programs into the airborne processor.

4) Conduct static tests of the completed system in the integration laboratory and low dynamics tests in a mobile van.

5) Conduct flight tests in a helicopter test aircraft to evaluate system performance in the high vibration environment.

6) Assess the navigation system's performance by conducting extensive flight tests in a helicopter test aircraft on an instrumented range.

The aim of Phase II is to realize in hardware the Phase I system design by constructing and developing through ground and flight testing a HINS Advanced Development Model (ADM). The system software developed in the Phase I study will be implemented in an airborne processor and the interfaces between the processor and sensors will be developed. The resulting system will be further developed through an extensive series of ground and flight tests.

At its completion, the project will have produced a fully developed and flight-validated navigation system ready for incorporation in the maritime helicopter. The project will also have enhanced the ability of the Department of National Defence to develop integrated navigation systems applicable to future Canadian Forces aircraft programs.

4.0 SYSTEM OVERVIEW

A typical structure of the HINS is illustrated in Figure 2. The primary system component is the integrated sensor unit which consists of all the selected navigation sensors (subsystems). They in general, can be classified into two main categories, dead reckoning and radio navigation systems. The dead reckoning (DR) systems are the inertial navigation system (INS),doppler radar and air data computer. These systems are self-contained and provide velocity information which can be integrated to obtain the position of the aircraft. Position estimates so obtained have a tendency to drift, slowly accumulating a large error over an extended flight. DR systems (except doppler) are generally non-radiating and non-jammable and hence attractive for operation in a hostile environment. On the other hand, radio navigation systems such as Omega, TACAN and GPS (Global Positioning System) rely upon radio signals transmitted from several external sources which make them vulnerable to enemy actions in any serious confrontation. These systems provide absolute position information with error characteristics which do not grow with the passage of time, because errors are not acumulated through an integration routine. In the short term however, radio aids excepting the GPS generally don't produce as stable and accurate a track as the DR systems.

In order to take advantage of the short-term stability of the DR system and the long term stability of the radio navigation system, one can apply the theory of optimal estimation in the form of a Kalman filter (K.F.) to combine all the available measurements and provide a statistically "optimal" estimate of the aircraft position, attitude and velocity, etc.

Another system component is the data processing computer which handles the workload of optimal integration, navigation and guidance functions. This means that large amounts of data can be handled at high rates, and the computer can be entrusted to carry out very sophisticated computations to provide much more reliable navigation and control output data.

To achieve the desired performance from HINS, the HINS processor must perform signal conditioning and sensor compensation functions, (Figure 3) and it must contain efficient K.F. integration algorithms, error control and performance monitoring routines. The signal conditioning function controls the sampling rate of incoming data and works as a prefilter to average the higher frequency data input. The sensor compensator compensates for predictable deterministic errors intrinsic to each sensor, and also provides compensation to remove the effects of unstable positive feedback in the GPS rate aiding loop. The performance monitoring routines serve as a watchdog to keep track of the overall system performance, and also provide CEP estimates to indicate the expected performance accuracy of the automated system.

The HINS integration filter will optimally weight and combine all available sensor data to estimate aircraft position, velocity and attitude with greater accuracy than available from any of the sensors individually. This means that mission requirements can be met by using sensors which are individually

less accurate and less expensive than would otherwise be required. In the event of subsystem failures, the system performance degradation will be graceful and the integrated system will automatically configure itself to the next optimum operational mode.

The use of standard Data Bus (MIL-STD-1553B) facilitates configuration selection and data interface of different brands and types of equipment, and allows integration of the HINS into future avionic suites. The use of a digital data bus provides a higher data flow capacity, self check on each data transmission and reduces susceptibility to electromagnetic interference. It also reduces weight and improves reliability because less wiring and fewer connectors are needed.

5.0 MISSION PROFILES

Two representative mission profiles for the maritime helicopter have been developed for use during the HINS simulation studies. The first profile is typical of an ASW sonobuoy mission while the second represents an ASW convoy screening mission. The HINS flight test program would seek to follow flight profiles similar to these:

5.1 ASW Sonobuoy Mission (see Figures 4 and 5)

(1) Mission Alert: (10 minutes)

- aircraft power turned on, helicopter-destroyer at speed of 18 knots. A contact is detected by the ship sensors at a range of 90 nm and a bearing at 90 degrees to the ship's course.

(2) Launch and Climb: (10 minutes)

- climb at a rate of 800-1000 ft/min. to a cruise altitude of 5000'.
- airspeed during climb 90-100 knots.
- course 235 degrees.

(3) Enroute/Cruise: (45 minutes)

- course 235 degrees.
- speed 140 knots, alt 5000', distance flown 100-110 nm.

(4) Sonobuoy Pattern Drop: (25 minutes)

- drop altitude 5000', speed 90 kts.
- rate 1/2 turns (1.5 deg/sec), 15 deg. bank angle.
- eight sonobuoys, 5 nm spacing.

(5) Oribit/Sonobuoy Monitor: (40 minutes)

- orbit altitude 5000', speed 70 kts, circular or racetrack pattern, may alternate direction.
- 10 nm long orbits, 15 minute circuits.

(6) Localize/Additional Sonobuoy Drop: (15 minutes)

- contact passes between sonobuoys A and B.
- helicopter descends to 500' at 1000 ft/min, flies 7 nm downrange and inserts two additional sonobuoys at C and D.
- orbit at 500 ft for 10 minutes to monitor sonobuoys A, B, C and D.

(7) Magnetic Anomaly Detector (MAD) Run: (10 minutes)

- use MAD to obtain a precise fix on the target by flying 1000' diameter ovals at 200' alt, always flying in the same direction.
- speed 90 kts, rate 1 turns (3 deg./sec), 22 degrees of bank.
- straight and level re-fly over high probability area.

(8) Attack: (5 minutes)

- helicopter descends to 150' and releases torpedo.

(9) Transit Back to Ship: (50 minutes)

- climb to 5000'..
- transit speed 140 kts.

5.2 Convoy Screening Mission (see Figure 6)

The maritime helicopter provides close-in ASW screening support to task force operation in open ocean. Dipping sonar is used to search a moving sector formed by a 60 deg. arc some 9-12 miles in front of the ships.

(1) Mission Alert: (10 minutes)

- aircraft power on.

(2) Launch/Climb/Enroute: (7 minutes)

- climb to 150' transit 10 nm to the search sector and begin first sonar dip at pt A.
- transit speed at 140 knots.

(3) Sonar Dipping: (15 minute cycle)

- this dipping sequence is repeated continuously for the duration of the 4 hr mission, with sonar dip pts. at A, B and C within the sector.
- cruise velocity between dips is 90 kts.
- cruise altitude between dips is 150 ft.

Dipping Operation:

(a) - 700 to 1000' before dip pt. helicopter turns into wind with rate 1/2 turn, 15 deg. bank, slows to 70 kts and descends to 150' alt.

(b) - transition to the hover at 50' altitude (max power setting, max vibration).
- lower sonar, hover at 50' for approximately 6 minutes.

(c) - retract sonar, move to next dip point at 70 kts. and 150' altitude.

(4) Repeat Sonar Dipping Cycle (as per (3) above), at B' and C'.

(5) Return to Ship: (10 minutes)

- velocity 140 kts.

6.0 CANDIDATE CONFIGURATION SELECTION

A wide variety of types and brands of navigation sensors can contribute to meeting the mission requirements of maritime helicopters. The following list of generic navigation subsystems is considered to have merit in the HINS application:

Global Positioning System (GPS),
Inertial Navigation System (INS),
Attitude and Heading Reference System (AHRS),
Doppler Radar,
TACAN,
Omega,
Air Data System,
Strapdown Magnetometer,
Loran,
Radar Altimeter.

However, many of the generic equipment catagories can be further subdivided. For example, in the INS category there are high and medium accuracy inertial equipments available. The inertial sensors can be mechanized either as a gimballed platform or a strapdown configuration. Strapdown configurations may use conventional or ring laser gyroscopes. Similarly, Doppler velocity sensors and Doppler navigation systems are available in the Doppler catagory.

While it is expected that Omega will become redundant once the GPS system is available, Omega and Loran have been included in this study as alternatives. Tacan serves a dual role in the maritime helicopter. Firstly, it can supply absolute navigation data from ground stations, and secondly a ship-based Tacan transmitter can be used by the helicopter for relative navigation and as a homing signal. Therefore, Tacan has been specified as required airframe equipment and been included in all configurations. The air data system and magnetometer* are also considered as part of the standard helicopter equipment and as such are not included in the calculation of weight volume and cost of the candidate sets. Furthermore, due to the relative inaccuracy of the barometric altitude information, a radar altimeter is considered necessary for low altitude hover. Therefore, a radar altimeter is also included in the evaluation of all candidate configurations.

In order to make an intelligent selection of a preferred HINS configuration, much information about candidate subsystems is required for evaluation. Amongst the characteristics for which information was sought are:

i) Performance (navigation parameter, accuracy, error characteristics)
ii) Cost
iii) Reliability (mean time between failure)
iv) Canadian content
v) Weight, volume and power
vi) Commonality with in-service equipment
vii) Suitability for helicopter environment

Other characteristics such as maintainability, logistics commonality with in-service equipment and helicopter suitability are being considered on a continuing basis throughout the study.

Solicitation for data was made to over 70 navigation manufacturers and suppliers. The information supplied was used to generate a data base of the characteristics shown above. A summary of the navigation sensor characteristics is given in Table 1. With few exceptions the cost figures are "budgetary" based on complete installations including mounts and control display units. The GPS subsystem cost data are particularly questionable because of the state of their developments. Most of the GPS quotes were based on expected production unit prices.

6.1 Evaluations Using Integrated System Evaluation Program (ISEP)

A computer-aided procedure called ISEP was developed by the Phase I contractor, Honeywell Ltd., to systematically develop candidate configurations. ISEP generates a manageable number of configurations, lists the system characteristics and calculates the predicted performance of the integrated system. The many candidate configurations can then be quickly reviewed by the designer and a subset selected for detailed simulation and analysis. A block diagram representation of ISEP is shown in Figure 7. The simplified error models for all the sensors considered are described in Table 2. ISEP uses a simplified set of sensor error models in a covariance analysis routine to generate performance predictions for the HINS configurations. Overly simplified INS and AHRS error models were used to coarsely represent the error characteristics of medium accuracy INS and LR-80 type subsystems, respectively.

The aim of the HINS project is to determine the optimum sensor configuration which satisfies the performance requirements. This does not necessarily mean that the optimum configuration will consist of those sensors which provide the highest level of accuracy. The trade-off of system accuracy with cost, reliability, weight and size must be considered during the equipment selection process. For example in the selection of either an INS or AHRS, the ISEP can be effectively used to determine whether an AHRS can satisfy the mission requirements or the improved accuracy to be gained from using a more accurate INS is justified in light of the associated cost, reliability and weight penalty. In addition, due to the lower cost of running the ISEP than that of a detailed generalized covariance analysis program, significant insight can be gained by using ISEP to identify critical design elements that require special attention during the detailed simulation and analysis portion of the project.

Some of the issues which can be initially addressed using ISEP are

i) the benefits of doppler aiding,
ii) the effects of initialization errors,
iii) mission profile sensitivity, and
iv) the benefit of external aiding.

There are five main sets of data which ISEP requires to generate the candidate configurations:

a) Mission requirements - performance specifications
b) Sensor characteristics - cost, MTBF, weight, etc.
c) Mission profiles - used to generate trajectories
d) Sensor error models
e) Selection criteria - weight or volume limits, etc.

Approximately 120 candidate configuration were generated by ISEP. After eliminating those configurations which were only small variations from another configuration, there were 45 configurations remaining for the designers to review.

6.2 ISEP Evaluation Results

The ISEP output data for a sample candidate configuration is shown in Table 3. The candidate HINS performance was evaluated by ISEP for both the aided mode of operation (externally referenced sensors such as GPS or Omega were contributing to the solution) and the unaided mode. It should be noted that unaided mode performance was evaluated assuming the availability and use of the Doppler velocity sensor.

The figure of merit for the unaided performance is the least square fit of the radial position error rate in nautical miles per hour (95%). The figures of merit for the aided performance are the maximum and average radial position error in nautical miles (95%). In all cases the calculated figures of merit are for two dimensional (level axes only) position errors. The performance goals for HINS are 1.5 nautical miles per hour (95%) for the radial position error rate during unaided operation and 2.0 nautical miles (95%) for the bounded radial position error with external aiding.

The ISEP simulation results are of course dependent on the types of mission profiles considered. There is significant uncertainty regarding how closely the performance predictions using these simplified models match the true performance of the system under consideration. Therefore, it was felt necessary to generate pessimistic as well as optimistic ISEP performance results in addition to the nominal results. In this simplified performance evaluation, the Doppler and Omega errors are represented by three different sets of parameters as shown in Table 4 to provide us with the envelope of performance results. The pessimistic values were used to represent reduced-order suboptimal filter design and the usage of poorer quality sensors. The optimistic values were selected to represent the limit of achievable performance.

In addition, it was found that the performance results of both the unaided and aided configurations, depend very much on the condition of initial alignment. Therefore, three different sets of initialization conditions (Table 5) were assumed to represent the ground fine alignment, shipboard fine alignment and in-air alignment.

The initial conditions specified for shipboard fine alignment assumed much poorer initial velocity and position accuracy figures than those specified for ground fine alignment.

AHRS vs. INS

The ISEP results presented in Tables 6-8 consistently indicate that the low accuracy AHRS integrated with a doppler performs drastically superior to the AHRS operating alone. The ISEP plots of 95% position error versus time for the INS/Doppler and AHRS/Doppler configurations are shown in Figures 8-9. The performances of these two configurations with initial ground fine alignment are evaluated against the convoy screening and Sonobuoy mission profiles. As shown in these figures, the nominal performance plot is bounded by the pessimistic and optimistic plots.

For the shorter duration convoy mission, the AHRS/Doppler unaided performance would in the most pessimistic case of ground fine alignment (Figure 8) be only 20% worse than the corresponding INS/Doppler performance. Whereas the more optimistic performance prediction of AHRS/Doppler is about 6% worse than the respective INS/Doppler prediction. As illustrated in Figure 8, the helicopter hovering at very low speed would enable the Kalman filter to significantly reduce the integrated system position error as well as the position error growth. In the cases of shipboard and in-air alignments, the AHRS/Doppler results are approximately 2.5% - 10% worse than the respective INS/Doppler performances for all three sets of doppler design parameters.

However, for the lengthier sonobuoy mission, the AHRS/Doppler unaided performance would in the most pessimistic case of ground fine alignment (Figure 9) be more than 2 to 1 worse than the respective INS/Doppler. Whereas the more optimistic performance prediction of AHRS/Doppler with ground fine alignment is about 60% worse than the respective INS/Doppler system. In the cases of shipboard and in-air alignments, the AHRS/Doppler results are approximately 30-90% worse than the respective INS/Doppler performances.

In general, the unaided performance of INS/Doppler would satisfy the 95% design goal of 1.5 NM/HR except in the cases of in-air alignment and short convoy mission with shipboard fine alignment. Whereas, the unaided performance of AHRS/Doppler would only be able to marginally meet the 95% design goal of 1.5 nm/hr when it could appropriately conduct ground fine alignment.

In addition, the comparison of Table 7 with Table 6 indicates that the unaided performances of Table 7 are drastically worse than those of Table 6 due to poorer initialization conditions specified for velocity and position, which means that in order to meet the 95% design goal of radial error rate, it might be necessary to provide not only precise attitude, but also position and velocity data transfer alignment. The availability of shipborne GPS/SINS output data for transfer alignment to the HINS would very effectively help both the INS/Doppler and AHRS/Doppler to meet the desired design goal. With the provision of GPS/SINS data for initial fine alignment, the expected performance of the unaided systems would not be too far off from that predicted in Table 6.

Furthermore, it should be noted that for INS/Doppler, the three sets of doppler design parameters have marginal effects on its performance accuracy. However, the performance results of AHRS/Doppler are extremely sensitive to the selection of doppler design parameters. This implies that to optimize the performance of the AHRS/Doppler integrated system, the filter design should be subject to much more stringent performance and sensitivity analyses than the design of the INS/Doppler system.

GPS vs. Omega

The extremely good accuracy available from GPS means that there is little doubt that it will be chosen over Omega as the primary external aid for the HINS, if all other factors are equal. However, considerations such as GPS availability and total ownership cost mean that the Omega cannot be discarded as a viable alternative. For example, Omega systems which are currently in the inventory could be included in the HINS as an inexpensive near-term substitute for GPS. For these reasons as well as the intention to have a more complete phase I study, Omega has been included in the HINS candidate configurations. This will assist the designer to determine whether the performance requirements can be met using Omega instead of GPS.

The results of Tables 9-11 indicate that the INS/Doppler/Ω would easily satisfy the 95% design goal of 2 nm position error except for in-air alignment. But the AHRS/Doppler/Ω would only be able to meet the 95% design goal of 2 nm position error for the short duration convoy mission with both ground and shipboard fine alignments.

For the convoy mission, the AHRS/Doppler/Ω would almost be as good as the INS/Doppler/Ω. This is consistent with the unaided performance result, since for the shorter mission, the effects of Ω long term stability are yet to be manifested.

For the lengthier sonobuoy mission, the AHRS/Doppler/Ω would in the most pessimistic case of ground fine alignment (Figure 10) be about 60% worse than the corresponding INS system. Whereas the more optimistic performance prediction of AHRS/Doppler/Ω is only about 28% worse than the respective INS system. In the cases of shipboard and in-air alignments, the AHRS/Doppler/Ω results are approximately 8-35% worse than the respective INS/Doppler/Ω results.

As a whole, the ISEP results indicate that all configurations considered can fully satisfy the weight and cost objectives, and they indicate that if AHRS were to be selected as a HINS subsystem rather than the more accurate and costly INS, it is unlikely that the Omega subsystem could provide accurate enough external references to meet the maritime warfare helicopter performance requirements. Even if INS were to be chosen, it is doubtful that the INS/Doppler and INS/Doppler/Ω configurations with in-air alignment would meet the performance requirements. Based on the ISEP results, one should have preceeded with the selection of AHRS/Doppler/GPS and totally discarded Omega and INS from further design consideration.

However, in order to have a more complete Phase I study and to ascertain the selection of the best integration configuration for HINS, it was felt necessary to proceed with more detailed covariance analysis and Monte Carlo simulations on three candidate configurations. They are catogorized as low, medium and high options, roughly assigned based upon their cost and performance as follows:

Low Option Configuration

AHRS/Doppler/Omega

Medium Option Configuration

AHRS/Doppler/GPS
INS/Doppler/Omega

High Option Configuration

INS/Doppler/GPS

By maintaining a broad scope for this study, one of the expected benefits is the ability to demonstrate a range of available performance levels and their associated cost. The results of the subsequent detailed study can also be compared to the ISEP results to determine the credibility of ISEP performance prediction using a simplified error model covariance analysis program.

7.0 SIMULATION AND PERFORMANCE ANALYSIS

In order to properly analyse the performance of various candidate configurations, a versatile simulation package is being developed. Since the fidelity and performance prediction of these configurations are of prime importance, a substantial portion of this development project was expended generating complete and accurate error models. These activities are described as follows:

a) Generate the navigation sensor error models.
b) Develop the integration algorithms to blend the sensor outputs.
c) Generate a set of mission profiles to be used for trajectory generation.
d) Develop control and display software.
e) Develop diagnostic software to detect sensor failure.

Covariance analysis techniques are used as the primary design analysis and performance assessment tool. The system performance estimates generated by covariance analysis will be verified using Monte Carlo simulation techniques.

7.1 Subsystem Modelling

Detailed simulation and error models for all the relevant navigation equipment and environmental disturbances (wind, sea currents) have been developed for the performance evaluation and sensitivity analysis.

Highlights of the sensor error models are listed below:

a. INS/IRS/AHRS

- error sources modelled
  - the following errors will be treated as random constants or slowly varying (Gauss-Markov) processes:

Gyro	Accelerometer
G^2-Drift Coefficient	
Bias Drift	Bias Error
Scale Factor Error	Scale Factor Error
Input Axis Misalignment	IA Misalignment
G-Sensitive Drift Coeff.	Vibration Induced Bias

  - the remaining errors will be treated in the following manner:

    Gyro Random Drift (Random Walk)
    Gyro Attitude Random Walk (White Noise Drift)
    Gyro and Accel Turn-on Transients (Exponential decay)

- modify the parameters associated with these errors to suit AHRS, Ring-laser, Platform or Strapdown system under consideration.

b. GPS

- 2 Channel receiver used
- accept the three dimensional position and velocity outputs and directly feed into the filter

c. Doppler Radar

- error sources modelled

- Speed offset error (Bias)
- Doppler radar error (Scale Factor)
- Doppler fluctuation error (White Noise)
- Sea current error (area correlation)
- Wind blown water speed error

d. Omega

- error sources modelled
  - Spatially correlated error
  - 24 hour periodic error

    Long-term errors

  - Short term temporally correlated error
  - Uncorrelated error

    Short-term errors

- phase measurements from 4 Omega stations used to generate 3 lines of position (LOP)

e. Radar Altimeter

- primary overwater error source is system scale factor (Bias)
- altitude variations due to terrain height makes rad. alt. unuseable for vertical channel stabilization. Used for hover stabilization only.

f. Magnetometer

- error sources modelled
  - Heading dependent error (Periodic)
  - Instrument bias and installation alignment error (Bias)
  - magnetometer fluctuation error (White noise).

g. Air Data

- barometric altitude errors
  - altitude variation in constant pressure surface (1st order Gauss-Markov process)
  - temperature dependent scale factor
  - static pressure measurement error
  - instrument error
- air speed errrors due to wind speed slowly varying with altitude changes (time and distance correlated process)

h. Tacan

- error sources modelled
  - time delay (Bias)
  - calibration (Bias)
  - random ranging errors (Random Bias)
  - bearing error (Random Bias and correlated error)

8.0 INTEGRATION SOFTWARE DESIGN AND CONFIGURATION ANALYSIS

8.1 Integration Software Design

The HINS processor is responsible for performing a wide variety of sensor integration functions. These functions can naturally be divided into two groups: the sensor blending function and auxiliary functions such as failure detection. The principle function of a blending algorithm is to properly select the gains used to weight information available from the navigation sensors. The goal of sensor blending is to achieve a composite result which is superior to that obtainable from any single source.

Over the last two decades, an ever increasing proportion of multi-sensor blending problems have been solved using Kalman filters. The principle advantage of the Kalman filter over other blending schemes is that it continuously calculates the appropriate gains to weight the sensor information. This feature is important in navigation problems due to the continually time varying accuracy exhibited by many of the navigation sensors. However, there is a price to be paid for the improved performance of the Kalman filter and this is the significant computational burden placed upon the airborne computer. The computer limitations are addressed by consideration of suboptimal, or "reduced order" filters. This approach eliminates certain states that are not critical to the solution or combines their effects with other states.

8.2 Filter Algorithm

The filtering algorithm implemented in this study is Bierman's U-D factorized Kalman filter. This algorithm avoids the explicit computation of the estimation error covariance matrix P:

$$P = E[(\underline{x} - \hat{\underline{x}})(\underline{x} - \hat{\underline{x}})^T]$$

where $\hat{x}$ is the estimated state and E[.] denotes the expectation operator. Instead, P is propagated in terms of its factors U and D:

$$P = UDU^T$$

where U is a unit upper triangular matrix and D is a diagonal matrix.

The U-D algorithm is efficient and provides significant advantages in numerical stability and precision. Specifically, the factorization of P provides an effective doubling in computer word length in covariance-related calculations, and avoids filter divergence problems which can arise in more conventional filter mechanizations due to loss of the positive (semi-) definite property of P through the accumulation of round-off and truncation errors.

8.3 Preliminary Kalman Filter State Selection

The preliminary selection of the Kalman filter for one of the candidate configurations is shown in Table 12. New states may be added or states deleted as indicated by the analysis of the simulation results.

8.4 Inputs to Filter

The inputs to the filter from the various navigation sensors are indicated below. Most of these input data are derived from the error models used in the simulation.

From INS: Aircraft pitch, roll, heading; wander azimuth velocities; latitude, longitude and wander angle

From Doppler: Aircraft body axis velocity components

From GPS: Latitude, longitude, altitude and velocities

From Air Data: Airspeed

From Tacan: Range and bearing, Tacan ground station coordinates

8.5 Outputs of Filter

The outputs of the Kalman filter are indicated below. This information will be displayed to the operator and possibly supplied to other aircraft systems and used in the generation of tactical plots.

Aircraft State: Body attitude, body velocity, latitude, longitude, altitude and a measure of system accuracy.

Other Data: Groundspeed, ocean currents, wind direction and magnitude, failure indications (based upon residual tests)

8.6 Configuration Analysis

There are many more system characteristics, besides the system accuracy, that are necessary for the navigation system designer to determine if all the requirements of the intended user are to be satisfied. Some of these characteristics and issues which will be addressed as part of the analysis of the candidate configurations are:

a) Failure Detection/Isolation - The inherent redundancy of multi-sensor navigation systems allows a designer to detect and isolate a failed sensor subsystem.

b) System Reconfiguration - Given that a failure has occurred, been detected and isolated, a design goal is to automatically reconfigure the system to remove that sensor from all navigation calculations.

c) Navigation Display, Operator Interface - The navigation data (position, velocity and attitude) are available for display purposes directly from the filter estimates. Additional information of use to the operator such as an estimate of the system error, and system status (failures) can also be built into the integration algorithms. An operator override feature is required to allow for entry of initial position, waypoints and navigation updates.

d) Degraded Performance - Estimates of the system performance at various levels of degradation are to be determined. This information can be obtained by using the simulation tools developed for HINS and performing a study in accordance with a failure tree such as is shown in Figure 11.

e) Alignment - Algorithms are required to allow at-sea and in-flight alignments of the inertial systems.

f) System Architecture - The system configuration definition will also address the development of the system architecture. The elements of the architecture that will be studied include communication formats and controls, operating principles and data bus design.

g) Processor Characteristics - Since the integration software must eventually be transferred to an airborne processor, estimates of the required processor capabilities are needed. These include:

- Instruction set
- Processor speed
- Memory capacity
- Floating point capability
- Weight, volume, power

8.7 Preferred Configuration Selection

One of the principle products of the Phase I study will be the identification of an optimum navigation system with specific sensor subsystems and the algorithms used to perform their integration. The criteria against which the selection of the preferred design will be made have been described earlier in this paper. It should be noted that the preferred configuration that is selected will be a set of "generic" sensor subsystems. A specific manufacturer or model type will not be designated, rather a choice of equipments which may be used in each generic catagory will be indicated.

9.0 REMAINING ACTIVITY

Remaining Phase I activity will focus on the following tasks:

(a) Completion of preliminary integration algorithm design for the four candidate system configurations.

(b) Evaluation of predicted performance in the baseline (non-degraded) and degraded modes of operation for each configuration over a selection of representative mission profiles. This evaluation will be conducted using both covariance analysis and simulation techniques.

(c) Development of software to perform the failure detection/isolation/reconfiguration function for each of the configurations.

(d) Examination of cost/weight/volume/performance/reliability tradeoffs for the candidate configurations leading to the recommendation of a preferred system configuration. This recommendation will also address the question of the use of embedded versus dedicated processing to implement the HINS integration algorithm.

(e) Development of a preliminary test plan for the Phase II laboratory/ground-mobile/flight test activities.

10.0 CONCLUSIONS

The objective of the HINS project is to design, develop and test an integrated navigation system for the maritime ASW helicopter. By investing a significant portion of the development effort in modelling and simulation studies the performance of the candidate configurations can be identified. Then tradeoffs can be made between the performance and the other system characteristics such as cost, reliability and size. The system designer and the customer can make the selection of the "optimum" design using performance estimates which have been generated by proven research techniques. The approach taken to the HINS development work is a systematic design process which will have application to the development of all types of integrated navigation systems.

REFERENCES

[1] Bierman, G.J., "Factorization Methods for Discrete Sequential Estimation", Academic Press, N.Y., 1977.

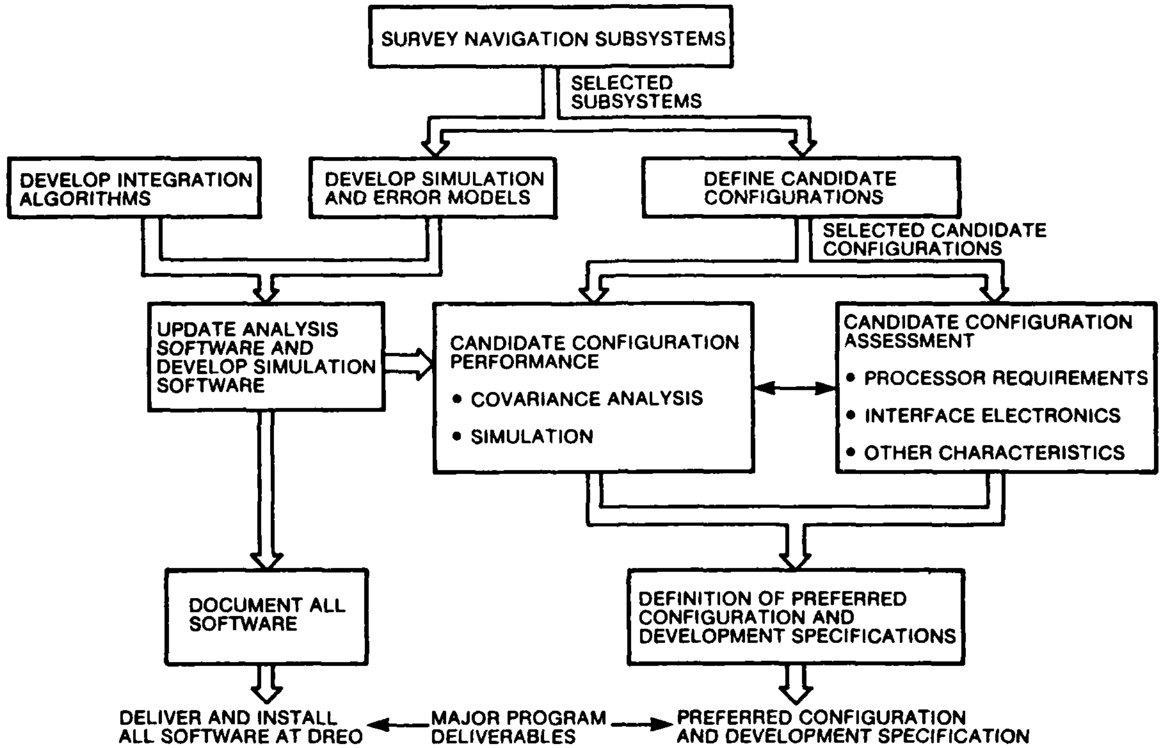

FIGURE 1: HINS PHASE I STUDY PLAN

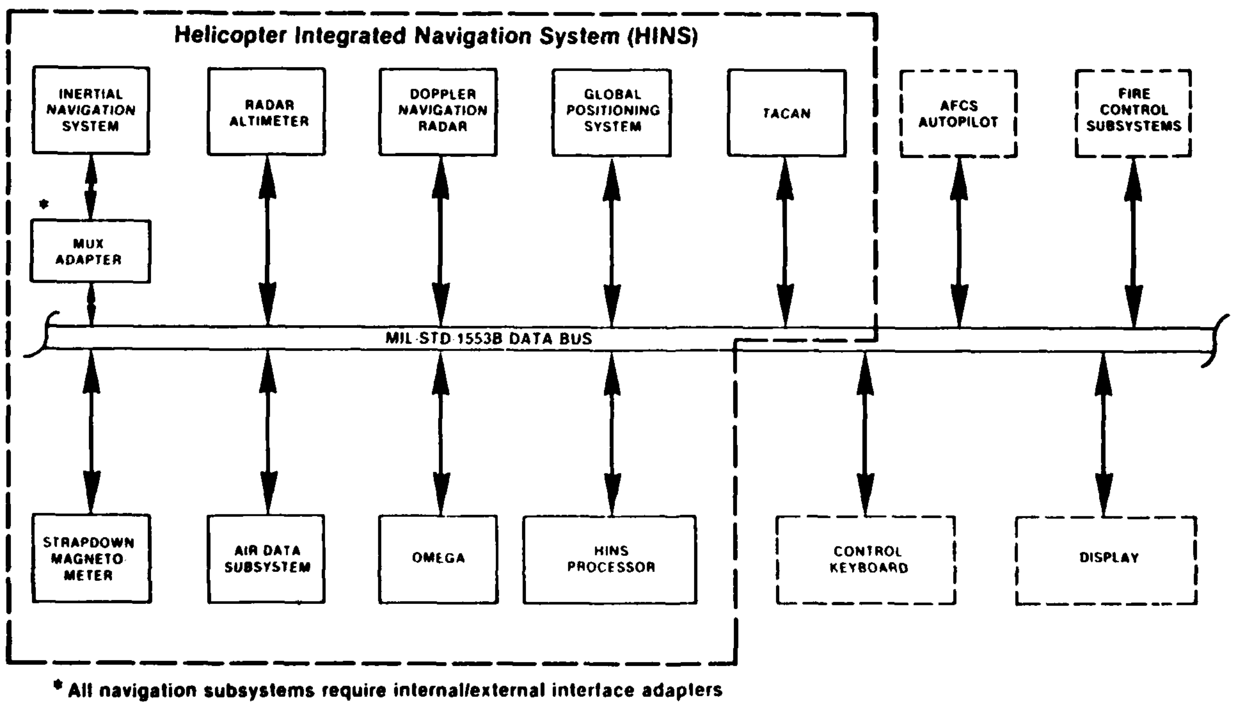

FIGURE 2: TYPICAL HINS CANDIDATE CONFIGURATION

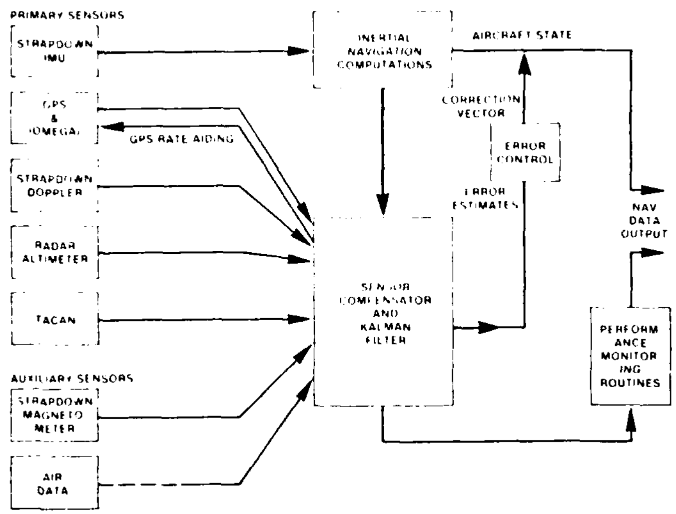

FIGURE 3: HINS INTEGRATION FUNCTIONS

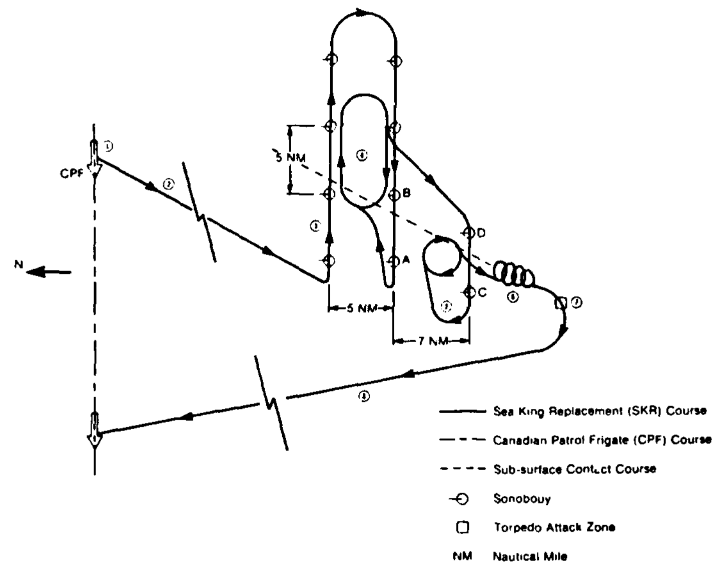

FIGURE 4: PLAN VIEW OF SONOBUOY MISSION

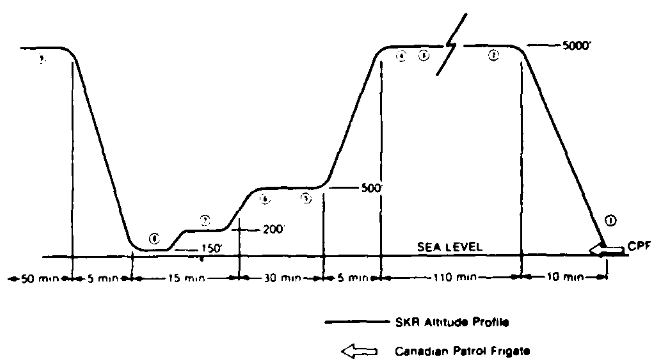

FIGURE 5: ALTITUDE PROFILE FOR SONOBUOY MISSION

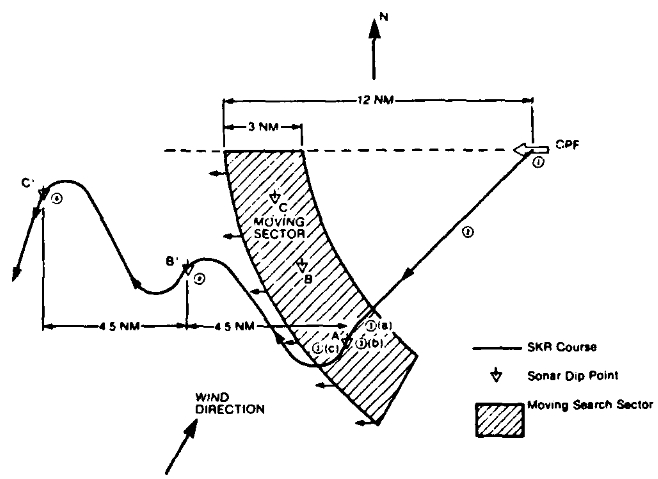

FIGURE 6: PLAN VIEW OF CONVOY SCREENING MISSION

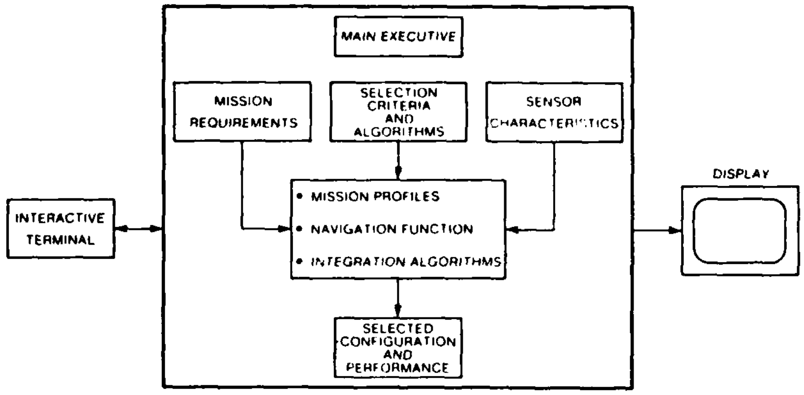

FIGURE 7: INTEGRATED SYSTEM EVALUATION PROGRAM

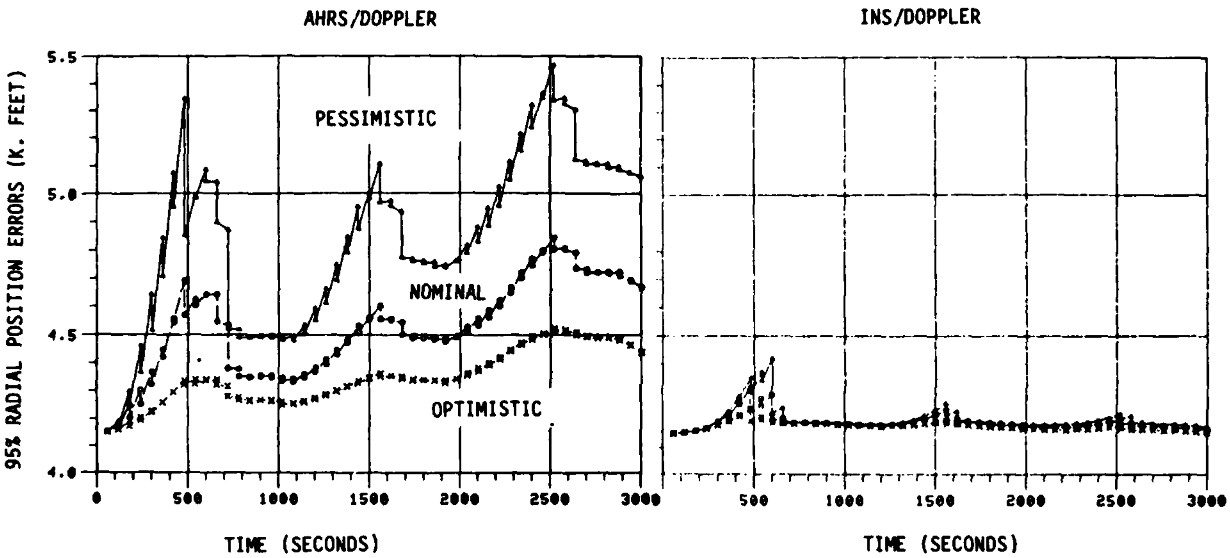

FIGURE 8: UNAIDED NAV SYSTEM PERFORMANCE WITH GROUND FINE ALIGNMENT (CONVOY MISSION)

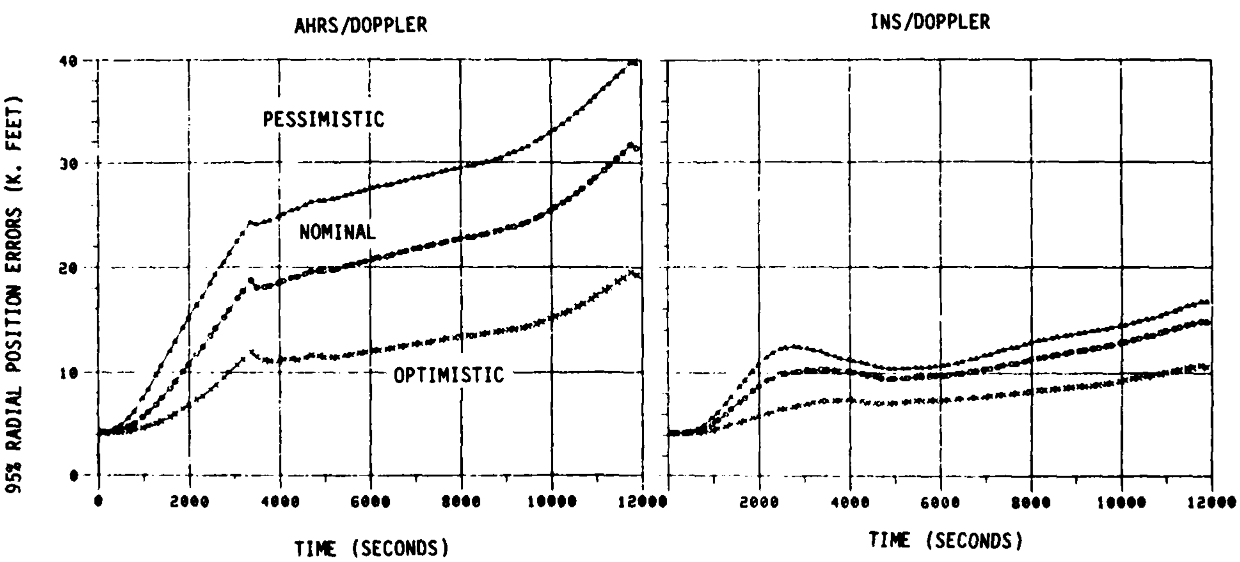

FIGURE 9: UNAIDED NAV SYSTEM PERFORMANCE WITH GROUND FINE ALIGNMENT (SONOBUOY MISSION)

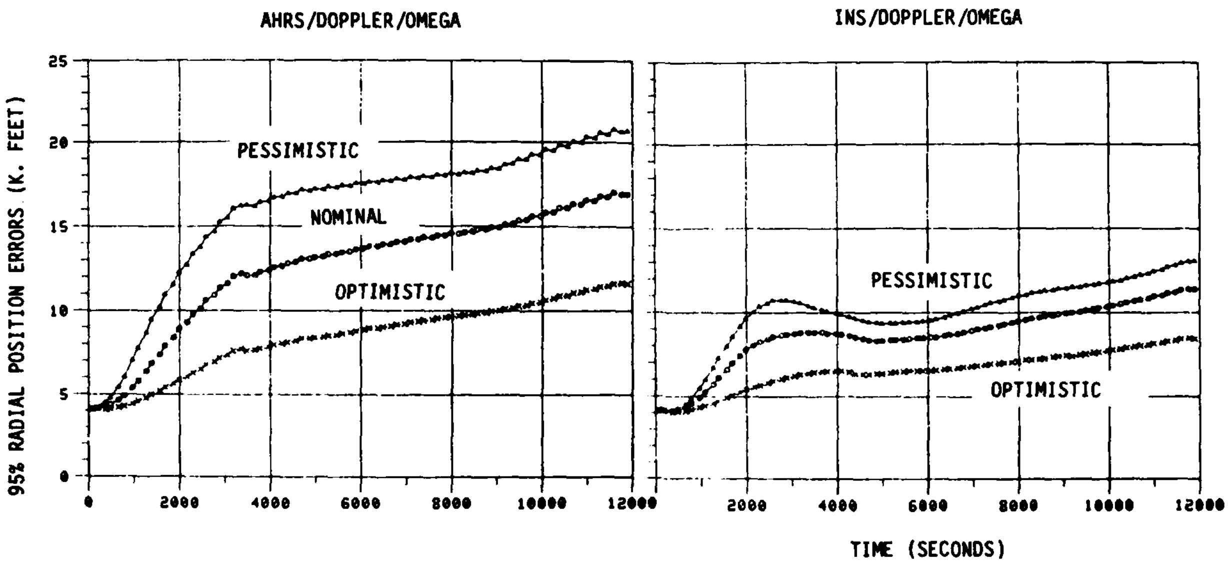

FIGURE 10: AIDED NAV SYSTEM PERFORMANCE WITH GROUND FINE ALIGNMENT (SONOBUOY MISSION)

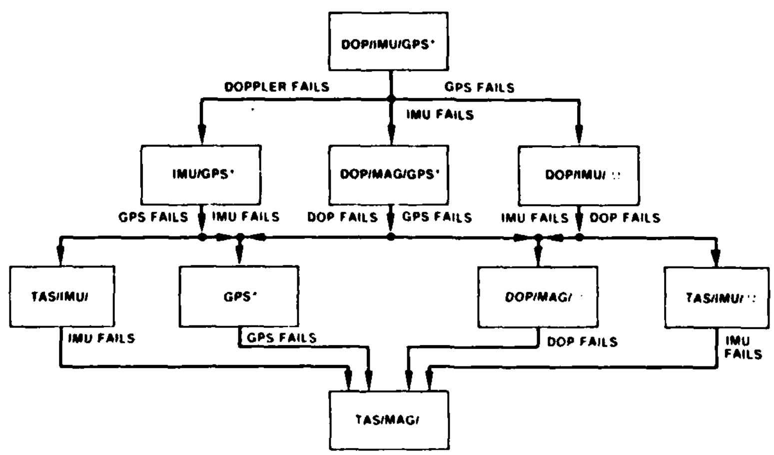

FIGURE 11: HINS FAILURE DIAGRAM

TABLE 1 - NAVIGATION SUMMARY CHARACTERISTICS

SUB SYSTEM	NO. OF SOURCES CONSIDERED	ACCURACY	INP/OP	MTBF HRS	VOL CU FT	WT. LB	PWR W/VA	(US$ 83) COST K$
AHRS	4	0.25°-1° AIDED	MIL-1553B	1200-7800	0.43-0.56	16-26	67-13	22-55
DOPPLER RADAR	1	0.1-0.3% VELOCITY	MIL-1553B	2100-4200	0.35-0.43	19-35	40-80	45-55
GPS	4	42-48', 0.3-2.1 FT/SEC	MIL-1553B	1000-5000	0.35-1.2	23-52	98-300	50 MIN
INS	10	0.6-2.0 NMPH	MIL-1553B	500-2500	0.6-1.1	35-60	110-300+	86-180
IRS	2	2 NM/HR	ARINC DIG'L	5000	0.6-0.7	44	90-110	105-115
DOPPLER INS	1	0.3% DIST. TRAVELED	MIL-1553B	800	0.68	36	135	-
INS OMEGA	1	3 NM, 0.25°-1°	ARINC DIG'L	5000	0.7 +	35	150	70
LORAN C	5	0.1-0.2 NM	ARINC	2700	0.22-0.42	4.5.12.	25-60	5-20
MAGNET OMETER	1	1°	ANALOGUE	-	0.005	0.4	1.3	1.7
OMEGA	10	2-4 NM	ARINC DIG'L	2000-5000	0.27-0.87	13-28	40-112	13-50
RADAR ALTIMETER	10	2-7% ALT	ARINC and 1553B	1000-8200	0.05-0.2	4-11	25-100	5-16
TACAN	6	0.1-0.15 NM, 1°	MIL-1553B	1000-1500	0.37-0.85	29-54	125-300	32-40
AIR DATA	2	2-4 KTS	ARINC and 1553B	3000 +	0.45 +	9-14	30-36	-
AIRBORNE PROCESSOR	4	200-850 KOPS	MIL-1750A	6000 +	0.9 TYPL	30-60	20+	-

TABLE 2 - ISEP SIMPLIFIED ERROR MODELS

SENSOR	DESCRIPTION	PER AXIS RMS (1σ)	CORRELATED TIME/DISTANCE
GPS	3 Dimensional Position Fix Errors 3 Dimensional Velocity Fix Errors	50 ft 0.3 ft/sec	
OMEGA	2 Orthogonal Position Errors	0.8 - 1.1 nm	1.3 - 4 hrs
MAGNETOMETER	Heading Bias Error Heading Random Error	1° 0.1°	
LORAN	2 Position Errors	0.2 nm	
TACAN	Correlated Range Error Uncorrelated Noise Correlated Bearing Error Uncorrelated Bearing Noise	0.1 nm 100 ft 1° 0.1°	0.5 hr 0.02 hr
DOPPLER	3-Axis Total Velocity Error Effects Uncorrelated Error	0.5 - 1.5% 0.25 - 1.5%	0.5 - 3 hrs
RADAR ALTIMETER	Uncorrelated Error	2-5%	
AHRS	Gyro Drift Accelerometer Drift	0.5°/hr 500 μg	0.5 - 3 hrs 0.5 - 3 hrs
INS	Gyro Drift Gyro Random Walk Accelerometer Drift	0.0035°/hr $0.0015°/\sqrt{hr}$ 35 μg	0.5 - 3 hrs 0.5 - 3 hrs
WIND	Correlated Error Random Fluctuation	50 ft/sec 5 ft/sec	1 hr

TABLE 3 - MINS OUTPUT DATA

MANUFACT MODEL NO	TYPE CANADIAN	COST MTBF	VOLUME WEIGHT	POWER MS1553B	MEASURE 1 VALUE 1	MEASURE 2 VALUE 2	MEASURE 3 VALUE 3
Litton LTN-90	INS yes	105000 5000	.702 44.000	W110 No	gyb-d/shr .010	rw-d/shr .005	ach-mig 85.00
Marconi Mar-2?h	GPS yes	50000 Est 1500	.647 UK	W250 Yes	sep-ft 45.00	v-ft/sec .3000	
Honeywell APN-194	Rad-Alt No	11000 4500	.052 4.400	W25 No	alt-% 4.00		
Marconi APN-227	Dop-Rad Yes	50000 2900	.353 35.000	VA65 Yes	vx-% .170	vy-% .170	vz-% .100
Marconi AM-143	Air-Data UK	N/A 3000	.452 11.200	W30 N/A	alt-ft 25.00	as-knots 15.00	
Develco D-9200C	Mag-Net No	1700 0	.005 .380	W1.3 No	att-deg 1.00		

Total Cost = $217,700 ($US 83)
Canadian Content = 94%
Total Volume = 2.215 cu-ft
Total Weight = 95.0 lbs
System MTBF (series model) = 698.1 hrs

Mission Performance (95% radial position error)

Unaided Performance = 1.0987 nm/hr
Aided Performance (max) = 0.03640 nm
Aided Performance (avg) = 0.00940 nm

TABLE 4 - OMEGA AND D[illegible]LER SIMPLIFIED ERROR MODELS

	PESSIMISTIC		NOMINAL		OPTIMISTIC	
	PER AXIS RMS	CORRELATED TIME	PER AXIS RMS	COR. TIME	PER AXIS RMS	COR. TIME
Doppler Cor. Noise	1.5%	0.25 Hour	1.0%	0.5 Hour	0.5%	1.0 Hour
Doppler Uncor. Noise	1.0%		0.5%		0.25%	
Omega Cor. Noise	1.2 nm	1.3 Hour	1.0 nm	2.0 Hour	0.8 nm	4.0 Hour
Omega Uncor. Noise	400 M		300 M		200 M	

TABLE 5 - INITIALIZATION CONDITION

		GROUND FINE ALIGNMENT	SHIPBOARD FINE ALIGNMENT	IN-AIR ALIGNMENT
INS	ATTITUDE	0.1 mrad	0.1 mrad	10 mrad
	AZIMUTH	1 mrad	1 mrad	20 mrad
	VELOCITY	0.2 ft/sec	1.5 ft/sec	5 ft/sec
	POSITION	0.2 nm	0.54 nm	1 nm
AHRS	ATTITUDE	0.5 mrad	0.5 mrad	10 mrad
	AZIMUTH	1 mrad	1 mrad	20 mrad
	VELOCITY	0.2 ft/sec	1.5 ft/sec	30 ft/sec
	POSITION	0.2 nm	0.54 nm	1 nm

TABLE 6 - UNAIDED PERFORMANCE (95%) RADIAL ERROR RATE (WITH GROUND FINE ALIGNMENT)

CONVOY MISSION

	FREE INERTIAL	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	73.94 nm/hr	1.30 nm/hr	1.21 nm/hr	1.15 nm/hr
INS	3.50	1.09	1.09	1.09

SONOBUOY MISSION

	FREE INERTIAL	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	39.77 nm/hr	2.23 nm/hr	1.73 nm/hr	1.03 nm/hr
INS	1.35	.97	.86	.63

TABLE 7 - UNAIDED PERFORMANCE (95%) RADIAL ERROR RATE (WITH SHIPBOARD FINE ALIGNMENT)

CONVOY MISSION

	FREE INERTIAL	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	74.06 nm/hr	3.02 nm/hr	2.97 nm/hr	2.92 nm/hr
INS	4.51	2.86	2.86	2.84

SONOBUOY MISSION

	FREE INERTIAL	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	39.78 nm/hr	2.38 nm/hr	1.92 nm/hr	1.30 nm/hr
INS	1.58	1.26	1.16	.97

TABLE 8 - UNAIDED PERFORMANCE (95%) RADIAL ERROR RATE (WITH IN-AIR ALIGNMENT)

CONVOY MISSION

	PESSIMISTIC	OPTIMISTIC
AHRS	5.65 nm/hr	5.37 nm/hr
INS	5.15	5.00

SONOBUOY MISSION

	PESSIMISTIC	OPTIMISTIC
AHRS	2.93 nm/hr	1.86 nm/hr
INS	2.03	1.40

TABLE 9 - AIDED PERFORMANCE (95%) RADIAL ERROR (WITH GROUND FINE ALIGNMENT)

CONVOY MISSION

	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	.78 nm	.72 nm	.69 nm
INS	.68	.68	.66

SONOBUOY MISSION

	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	2.66 nm	2.09 nm	1.40 nm
INS	1.65	1.44	1.09

TABLE 10 - AIDED PERFORMANCE (95%) RADIAL ERROR (WITH SHIPBOARD FINE ALIGNMENT)

CONVOY MISSION

	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	1.73 nm	1.65 nm	1.56 nm
INS	1.67	1.62	1.53

SONOBUOY MISSION

	PESSIMISTIC	NOMINAL	OPTIMISTIC
AHRS	2.90 nm	2.41 nm	1.86 nm
INS	2.14	1.94	1.65

TABLE 11 - PERFORMANCE (95%) RADIAL POSITION ERROR (IN-AIR ALIGNMENT)

CONVOY MISSION

	PESSIMISTIC	OPTIMISTIC
AHRS	2.69 nm	2.22 nm
INS	2.60	2.17

SONOBUOY MISSION

	Pessimistic	Optimistic
AHRS	3.27 nm	2.25 nm
INS	2.85	2.08

TABLE 12 - KALMAN FILTER STATE SELECTION

NAVIGATION STATES	ERROR STATES	DISTURBAN...
3 Attitude Errors	3 Gyro Drift Rates	2 Sea Currents
2 Velocity Errors	2 Accel. Bias Errors	2 Wind States
2 Position Errors	8 Omega States (1 long term error state and 1 short-term error state for each of the four stations)	
	1 Doppler Along Track Error State 1 Doppler Across Track Error State	

High Accuracy Doppler Navigation employing optimum integrated Position Fix Information

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ABSTRACT

The basic accuracy limitations of today's Doppler navigation systems are mainly caused by sensor errors and calibration errors. The navigation error of a typical Doppler system equipped with a magnetic compass is therefore about 1 % to 2 % of the distance travelled. The presented approach for improved accuracy is based on the usually applied method of position fixing to reset the error at specific waypoints. The measured errors are used as an input to a Kalman Filter which estimates various error parameters and continuously compensates for the actual navigation error.

The paper presents details of the implemented Kalman Filter algorithm for optimum integration of position fix information in a Doppler navigation system, together with simulation results based on computer generated synthetic flight data and recorded real flight data.

INTRODUCTION

Doppler navigation is nowadays the most commonly used principle of self contained navigation in helicopters. This is because navigation based on the Doppler effect provides many features which are of particular importance to helicopters. Among these are properties such as self contained operation in remote areas without external aids, simple procedures for operation and maintenance, small size and weight and, after all, a relatively low cost.

A Doppler navigation system requires three major elements: A Doppler radar velocity sensor (DRVS), an aircraft heading and attitude reference and a suitable navigation computer including input/output capabilities.

Although the currently available DRVS provide a quite good accuracy, the performance of an entire system suffers from inaccuracies of the angle sensor subsystems as well as from calibration residual errors. For typical systems with heading supplied from the helicopter's magnetic compass with a "one sigma" error of one degree, the resulting navigation error is about 1 % to 2 % of the distance travelled. Systems of this medium accuracy have become a standard for helicopters in many military forces. However, the improved requirements for next generation military helicopters call for navigation systems with a total system error of less than 1 % of distance travelled.

There are several possible approaches under discussion which may meet these requirements. Among these are for instance satellite and strap down inertial navigation in stand alone configuration or integrated together with Doppler velocity sensors. But as these kinds of systems may propably not be made available at a reasonable cost in the very near future, there will still be a market for improved stand alone Doppler navigation systems.

Improved accuracy for Doppler navigation systems can be gained by either a more accurate DRVS, more accurate heading and attitude sensors and, what is the subject of this paper, by sophisticated signal processing. The signal processing approach as presented in the following, is based on optimal linear filtering. It utilizes a conventional Kalman Filter to provide an optimum linear estimate for significant system errors by means of a linear dynamic error model and additionally available navigation information from external measurements. Additional information can be provided by a simple operational method which does not require any extra sensors. This method is called position fixing and is commonly used in operating Doppler navigation and other kinds of dead reckoning navigation systems. Position fixing results in an error reset at specific waypoints with a priori known coordinates but the effective error mechanisms remain unchanged for standard Doppler navigation systems. The Kalman Filter, however, takes advantage of these external position error measurements in order to learn about the error behaviour.

The Kalman Filter for optimum estimation of errors inherent in Doppler navigation is derived in the following sections, starting with the mathematical treatment of the Doppler navigation principle and a subsequent error analysis which yields the analytical error models for the Kalman Filter.

PRINCIPLE OF DOPPLER NAVIGATION

Aircraft navigation based on the Doppler effect comprises the measurement of aircraft velocity by means of a Doppler radar, the measurement of aircraft direction and attitude by means of direction and attitude sensors and the suitable processing of these data to compute the actual aircraft position. The Doppler radar determines the aircraft velocity in direction of radar beams by extracting the Doppler shifts between radar signals which are transmitted from the aircraft and backscattered from the ground. Standard DRVS generally use four noncoplanar radar beams in a symmetrical arrangement as shown in Fig. 1. This provides a somewhat redundant basis for the determination of the aircraft velocity in an aircraft based perpendicular frame. To achieve this, the Doppler shifts f_{Di}, $i = 1,2,3,4$ optained from four beam directions are transformed on a minimum mean square error basis into velocity components v_x'', v_y'' and v_z'' according to the roll, pitch and vertical axis of the aircraft. In matrix notation, this transformation can be written as

$$\underline{v}'' = \underset{\sim}{T}_1 \cdot \underline{f}_D \quad , \qquad (1)$$

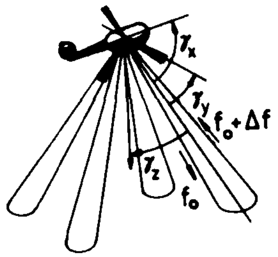


Figure 1: Radar beam arrangement in a 4 beam DRVS

where $\underline{f}_D$ and $\underline{v}''$ are vector notations of the Doppler shifts and velocity components respectively. The transformation matrix T_1 depends on the radar signal wavelength λ and the beam direction angles γ_x, γ_y, γ_z as shown in Fig. 1 and is given by

$$T_1 = \frac{\lambda}{8} \begin{bmatrix} \frac{1}{\cos\gamma_x} & \frac{1}{\cos\gamma_x} & -\frac{1}{\cos\gamma_x} & -\frac{1}{\cos\gamma_x} \\ -\frac{1}{\cos\gamma_y} & \frac{1}{\cos\gamma_y} & \frac{1}{\cos\gamma_y} & -\frac{1}{\cos\gamma_y} \\ \frac{1}{\cos\gamma_z} & \frac{1}{\cos\gamma_z} & \frac{1}{\cos\gamma_z} & \frac{1}{\cos\gamma_z} \end{bmatrix}$$

The aircraft based coordinate frame is then rotated by means of the vehicle roll, pitch and heading angles in order to determine the aircraft velocity components v_x, v_y, v_z in a north/east/down oriented frame.

The required coordinate frame rotation is performed in the following steps: A transformation into an earth based horizontal/vertical frame is carried out first, involving pitch and roll angles P and R of the aircraft. The velocity vector $\underline{v}'$ in this frame is determined as

$$\underline{v}' = \underset{\sim}{T}_2 \, \underline{v}'' , \qquad (3)$$

where the orthogonal transformation matrix $\underset{\sim}{T}_2$ is given by

$$\underset{\sim}{T}_2 = \begin{bmatrix} \cos P & \sin R \sin P & \cos R \sin P \\ 0 & \cos R & -\sin R \\ -\sin P & \sin R \cos P & \cos R \cos P \end{bmatrix} \qquad (4)$$

The horizontal/vertical frame is subsequently rotated in the horizontal plane by the amount of the aircraft orientation angle H with respect to the geographic north direction.

The resulting velocity vector $\underline{v} = \left[v_x, v_y, v_z\right]^T$ in north/east/down coordinates is then given by

$$\underline{v} = \underset{\sim}{T}_3 \, \underline{v}' = \underset{\sim}{T}_3 \, \underset{\sim}{T}_2 \, \underset{\sim}{T}_1 \, \underline{f}_D \qquad (5)$$

with the transformation matrix

$$\underset{\sim}{T}_3 = \begin{bmatrix} \cos H & -\sin H & 0 \\ \sin H & \cos H & 0 \\ 0 & 0 & 1 \end{bmatrix} \qquad (6)$$

The north and east velocity components v_x and v_y are finally integrated into north and east components of distance travelled from the point of departure.

The block diagram in Fig. 2 illustrates the entire Doppler navigation algorithm as it was described above.

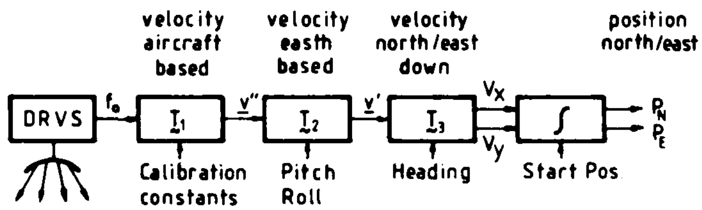


Figure 2: Doppler navigation functional block diagram

For the practical operation of a Doppler navigation system, only a few preflight data entries are required. Among these are for instance the starting point position which represents the initial condition for the integrator in Fig. 2 and the destination point position which can be used to compute steering information. Beyond this, additional waypoints are generally entered in order to operate the system between shorter distances.

This is advantageous because, as will be shown later, the errors in Doppler navigation are propagating with increasing distance and flight time, i. e. the indicated position becomes more and more inaccurate. This effect is demonstrated in Fig. 3, where the typical Doppler navigation error is shown for a straight flight at constant velocity. The upper curve shows the monotonously increasing error for a free running doppler navigation system, while the lower one is representing the error for the practical application of position fixing by means of waypoints with known coordinates. This results in resetting the error to zero each time a position fix is performed.

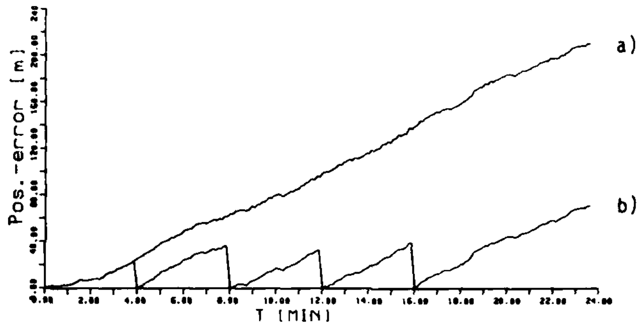


Figure 3: Doppler navigation position error for a straight flight
a) Free running Doppler navigation system
b) Doppler navigation employing position fixes

In order to reduce the overall error in Doppler navigation by means of an error estimation, the error propagation mechanisms have to be well understood and analysed carefully. This is carried out in the next section.

ERROR ANALYSIS AND MODELING

The errors inherent in Doppler navigation can be classified into the major categories of sensor errors, calibration errors and numerical errors.

Sensor errors come up from the DRVS and from the external angle reference subsystems. Calibration errors are present mainly because of antenna misalignment and numerical errors are due to word length effects within the digital navigation computer.

The individual errors are affecting different parts of the navigation algorithm. This is shown schematically in the functional block diagram in Fig. 4.

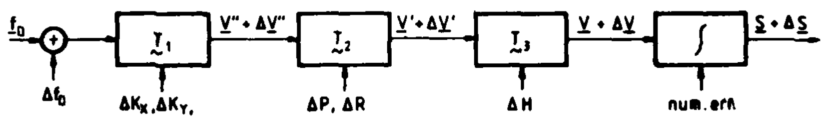


Figure 4: Errors in Doppler navigation

The Doppler fluctuation error vector $\Delta \underline{f}_D$ acts as a noise input to the entire navigation algorithm while the other errors are affecting the algorithm itself. The calibration errors which can be specified as two independant antenna misalignment quantities ΔK_x and ΔK_y are changing the $\underset{\sim}{T}_1$ transformation while the pitch and roll angle errors ΔP and ΔR result in incorrect transformations $\underset{\sim}{T}_2$ and $\underset{\sim}{T}_3$ respectively.

Together with the numerical errors which are present throughout the algorithm, all these error mechanisms end up in a total velocity error vector $\Delta \underline{v}$ which is finally integrated to give the total position error $\Delta \underline{s}$.

In order to develop a Kalman Filter for the estimation of this total error, the individual error sources have first to be characterized analytically. This can be done by means of physical error models and the analysis of measured data. For this purpose, a large number of test flights have been carried out and data was recorded for specific mission profils which allowed the separation and quantification of individual error sources. The test flights have been performed with a BO 105 helicopter and a navigation equipment as listed in Table 1.

Subsystem	Type
Doppler	AN/ASN 128
Heading Ref.	AN/ASN 89
Attitude Ref.	9000 C

Table 1: Navigation Equipment

The recorded flight data consists of the Doppler frequencies, pitch, roll and heading angles as well as time dependent present position values and position errors at waypoints. An example of recorded flight data is shown in Fig. 5. The error components that could be identified from the measured data are now considered in detail.

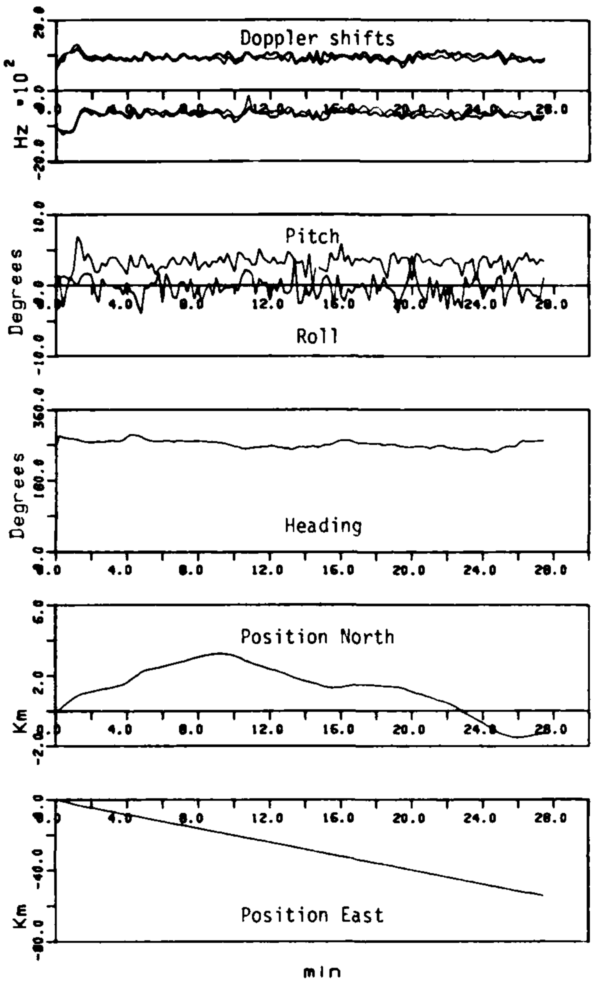


Figure 5: Example of recorded flight data

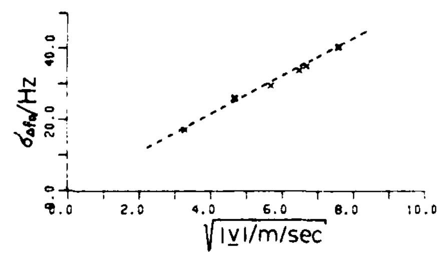


Figure 6: Standard deviation $\sigma_{\Delta f_D}$ of Doppler fluctuation error versus square root of the absolute aircraft velocity

a) Doppler fluctuation error

The Doppler frequencies as measured by the DRVS are output of a frequency tracker. They represent the center frequency of the narrowband Doppler spectrum which is fluctuating for instance because of the backscattering properties of the ground [1]. The fluctuation is correlated according to the tracker time constant and its standard deviation $\delta\Delta f_D$ can be shown to be proportional to the square root of the aircraft's absolute velocity [1]:

$$\sigma_{\Delta fD} = c \cdot \sqrt{|\underline{v}|} \tag{7}$$

The analysis of the measured data verifies the relation in eq. (7) with the constant factor $c \approx 5$ Hz / $\sqrt{m/sec}$. This can be seen from Fig. 6, where $\sigma\Delta f_D$ is plotted versus the square root of the velocity for several flights at different velocities. Fig. 7 shows an ensemble of normalized correlation functions for measured fluctuation errors taken from various flight samples. The correlation plots in Fig. 7 are justifying that these fluctuation errors may adequately be approximated by an exponentially correlated noise with the correlation time $T_{\Delta f} \approx 0.3$ sec taken as a mean value from Fig. 7. The error model is therefore represented by the differential equation

$$\dot{\Delta f}_D = -\frac{1}{T_{\Delta f_D}} \Delta f_D + n_{\Delta f} \,, \tag{8}$$

where $n_{\Delta f}$ is a forcing white noise with the spectral density

$$S_{n_{\Delta f}} = 2 \frac{c^2 |\underline{v}|}{T_{\Delta f}} \,. \tag{9}$$

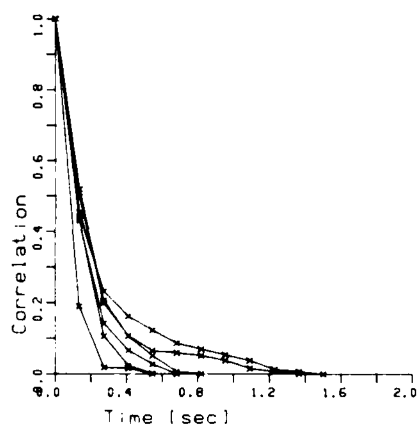


Figure 7: Ensemble of normalized correlation plots for measured Doppler fluctuation errors

b) Calibration errors

As already mentioned above, the calibration errors are mainly caused by antenna misalignment, i.e. by radar beam direction errors $\Delta\gamma_x$, $\Delta\gamma_y$ and $\Delta\gamma_z$ (Fig. 1). However, only two of these angle errors are linearly independent, so that for instance only $\Delta\gamma_x$ and $\Delta\gamma_y$ have to be considered. These errors are suitably represented by the normalized quantities

$$\frac{\Delta K_{x,y}}{K_{x,y}} = \frac{\cos\gamma_{x,y}}{\cos(\gamma_{x,y}+\Delta\gamma_{x,y})} - 1 \,, \tag{10}$$

where $K_{x,y}$ are abbreviated notations for elements in the T_1 transformation matrix:

$$K_{x,y} = \frac{\lambda}{8} \; \frac{1}{\cos\gamma_{x,y}} \quad .$$

The calibration errors can be stated to be offset errors, i.e. random constants with a zero derivative:

$$\left[\frac{\dot{\Delta K}_{x,y}}{K_{x,y}}\right] = 0 \tag{11}$$

This is confirmed by the measured data what is demonstrated in Fig. 8 where the calibration errors are depicted that have been derived from several test flights.

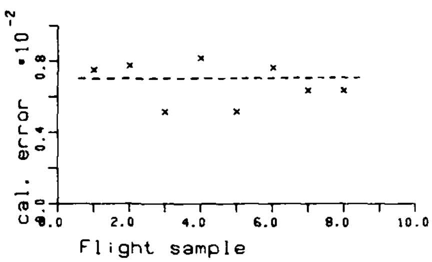


Figure 8: Relative calibration errors for several test flight samples

The average error $\Delta K_x/K_x$ which is effective along the aircraft's longitudinal axis can be identified from Fig. 8 as

$$\frac{\Delta K_x}{K_x} \approx -6.75 \cdot 10^{-3}$$

The corresponding alignment angle error $\Delta\gamma_x$ is about 0.1 degrees.

However, no noticeable error ΔK_y in aircraft lattitude or pitch axis could be identified even for flights in pure lattitude direction. This is important, because the ΔK_y error is effective only for a nonzero drift velocity and moreover it does not have a significant quantitative effect. Therefore, this error component can be neglected when modeling the calibration errors.

c) Pitch and roll angle errors

The vertical reference under consideration is composed of an averaging gravitational field sensor. The errors inherent in this device are partially manoever dependent but a long term statistical data analysis indicates that pitch and roll angle errors ΔP and ΔR may also be modeled by simple first order differential equations

$$\dot{\Delta P} = -\frac{1}{T_{\Delta P}} \cdot \Delta P + n_{\Delta P} \tag{12}$$

and

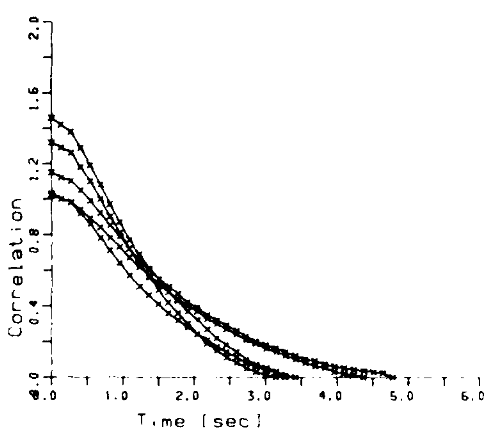
$$\dot{\Delta R} = -\frac{1}{T_{\Delta R}} \cdot \Delta R + n_{\Delta R} \tag{13}$$

The respective correlation times $T_{\Delta P}$, $T_{\Delta R}$ and spectral densities $S_{n_{\Delta P}}$, $S_{n_{\Delta R}}$ of the forcing white noise processes can be estimated from figs. 9 a, b, where measured correlation plots are shown for pitch and roll angle errors. The appropriate values for a good approximation are

$$T_{\Delta P} = 2 \text{ sec} \quad , \quad S_{n_{\Delta P}} = 2 \cdot 10^{-4} \text{ sec}^{-1}$$

and

$$T_{\Delta R} = 2 \text{ sec} \quad , \quad S_{n_{\Delta R}} = 5.6 \cdot 10^{-4} \text{ sec}^{-1} \quad .$$



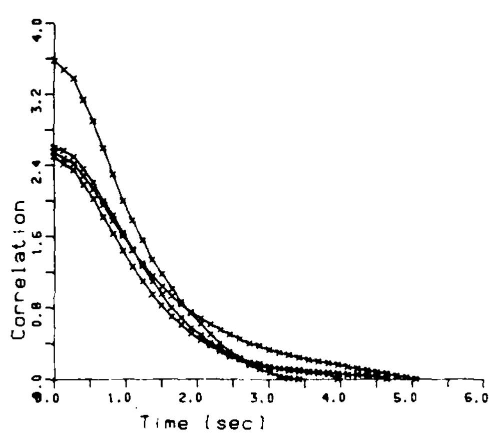


Figure 8: a) Pitch b) Roll
Ensemble of correlation plots for measured pitch and roll errors

d) Heading angle error

The most commonly used heading references in small aircrafts like helicopters are gyro stabilized magnetic compass sets. These devices show a very complex error behaviour which cannot be analysed and modeled easily. The resulting heading error further depends on external effects like magnetic aircraft components and unnormalities of the earth magnetic field. An adequately adapted error model therefore has to take into account error components like slowly varying sensor noise, magnetic variation errors and direction dependent magnetic deviation errors. The latter effects are generally even predominating the time varying effects and therefore the total heading error ΔH may be modeled as an offset error ΔH_o plus one and two cycle direction dependent errors [1] :

$$\Delta H = \Delta H_o + \Delta H_\alpha \sin H + \Delta H_\beta \cos H + \Delta H_\gamma \sin 2H + \Delta H_\delta \cos 2H \tag{14}$$

However, there is in general an opportunity to compensate for the individual error components in eq. (14), so that the residual error practically will depend on the goodness of calibration, i.e. it will be different from helicopter to helicopter.

Beyond this, the error components in eq. (14) are generally not observable in normal flight, i.e. they cannot be separated an therefore eq. (14) does not provide an appropriate error model for use in a Kalman Filter. It is a common practice to overcome such modeling problems by using a statistical error model in form of a random walk process [2] . The random walk model for the heading angle error can be written as

$$\Delta\dot{H} = n_{\Delta H} \quad , \tag{15}$$

where $n_{\Delta H}$ again is a forcing white noise process which is integrated to produce the heading angle error.

THE KALMAN FILTER APPROACH

A Kalman Filter by definition is an algorithm for estimating the state of a linear dynamic system which is corrupted by noise. The estimate is performed as a state vector extrapolation by means of a linear system model and after all, by additional external measurements which are used appropriately to update the extrapolation. For this purpose, the measured quantities must be linearly related to the state vector.

In the intended application of position fixing, the occasionally available measurements consist of north and east position errors. Therefore, the state space formulation of the linear dynamic system model implemented in the Kalman Filter has to contain these position errors as state variables.

A linear differential equation for the north and east position errors ΔN, ΔE can easily be found by differentiating the velocity vector $\underline{v}$ in eq. (5) with respect to the disturbed quantities:

$$\begin{aligned}\Delta\underline{v} = & \frac{\partial \underset{\sim}{T}_3}{\partial H} \underset{\sim}{T}_3^{-1} \underline{v}\, \Delta H + \underset{\sim}{T}_3 \frac{\partial \underset{\sim}{T}_2}{\partial P} \underset{\sim}{T}_2^{-1} \underset{\sim}{T}_3^{-1} \underline{v}\, \Delta P \\ & + \underset{\sim}{T}_3 \frac{\partial \underset{\sim}{T}_2}{\partial R} \underset{\sim}{T}_2^{-1} \underset{\sim}{T}_3^{-1} \underline{v}\, \Delta R + \underset{\sim}{T}_3 \underset{\sim}{T}_2 \frac{\partial \underset{\sim}{T}_1}{\partial K_x} K_x \underline{f}_D \frac{\Delta K_x}{K_x} \\ & + \underset{\sim}{T}_3 \underset{\sim}{T}_2 \underset{\sim}{T}_1 \Delta \underline{f}_D\end{aligned} \tag{16}$$

As the vertical velocity component is not of interest for areal navigation, the horizontal components of eq. (16) can directly serve as a suitable linear state space formulation of minimum dimension. This can be written as

$$\begin{bmatrix} \Delta N \\ \Delta E \end{bmatrix}^{\bullet} = \underset{\sim}{F}_o(t) \cdot \begin{bmatrix} \Delta N \\ \Delta E \end{bmatrix} + \underset{\sim}{G}_o(t) \cdot \underline{n}(t) \tag{17}$$

where $\underset{\sim}{F}_o(t)$ and $\underset{\sim}{G}_o(t)$ are the system matrix and noise input matrix respectively and n(t) is a white noise vector. However, the linear dynamic error model in eq. (17) will not be sufficient, because the involved error components do not represent white noise sources what is required for the Kalman Filter. Therefore, the state vector in eq. (17) has to be augmented by all these error quantities, which cannot be considered as white noise. This is achieved by means of the earlier derived differential equations for the characterization of the correlated noise sources. A completed state space formulation may therefore be written as

$$\begin{bmatrix} \Delta N \\ \Delta E \\ \Delta H \\ \Delta P \\ \Delta R \\ \Delta K_x/K_x \end{bmatrix}^{\bullet} = \begin{bmatrix} 0 & 0 & a_1 & b_1 & c_1 & d_1 \\ 0 & 0 & a_2 & b_2 & c_2 & d_2 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & -\frac{1}{T_{\Delta P}} & 0 & 0 \\ 0 & 0 & 0 & 0 & -\frac{1}{T_{\Delta R}} & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \end{bmatrix} \cdot \begin{bmatrix} \Delta N \\ \Delta E \\ \Delta H \\ \Delta P \\ \Delta R \\ \Delta K_x/K_x \end{bmatrix} + \begin{bmatrix} n_{\Delta N} \\ n_{\Delta E} \\ n_{\Delta H} \\ n_{\Delta P} \\ n_{\Delta R} \\ 0 \end{bmatrix} \tag{18}$$

where the calibration, pitch, roll and heading errors have been included as state variables and $n_{\Delta N}$, $n_{\Delta E}$ are horizontal velocity noise components which represent the Doppler fluctuation errors. The matrix elements a_i, b_i, c_i, d_i can be found by evaluating eq. (16).

Because of their weak correlation, the Doppler fluctuation errors are considered as white noise in order to keep the state vector dimension as small as possible. The danymic properties of this linear error model have been investigated by simulation using computer generated synthetic flight data that provided a continuous and perfect reference.

Fig. 10 shows simulation results for north and east position errors, exactly as they are occuring and as they are estimated by a Kalman Filter based on the error model in eq. (18).

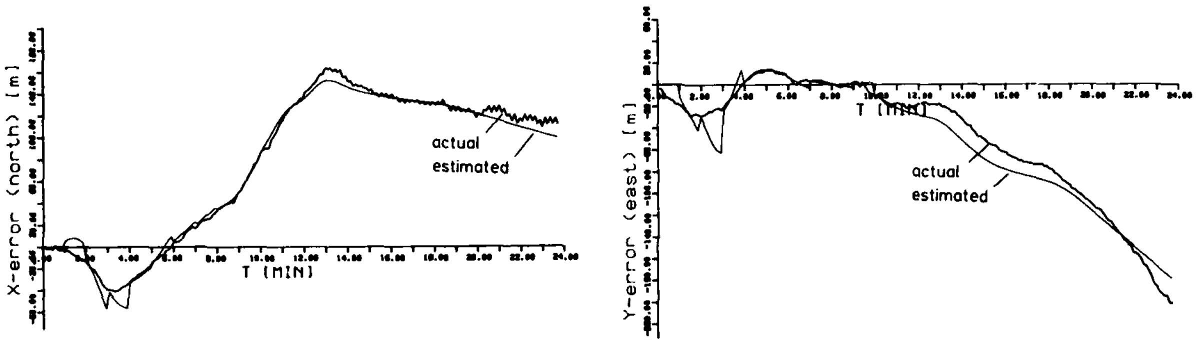


Figure 10: Doppler navigation errors in north and east, actual errors and errors estimated by the Kalman Filter

From Fig. 10 it can be seen that the actual and estimated errors agree very well after a learning period involving some position fixes at the beginning of the simulated flight.

The complete algorithm for Doppler navigation including Kalman Filter to integrate position fix information is summarized in Table 2 and an information flow diagram is shown in Fig. 11.

Discrete time:	$t = \nu T_A \rightarrow \nu$
Doppler Nav.:	$\underline{v}(\nu) = T_3(\nu) \cdot T_2(\nu) \cdot T_1 \cdot f_D(\nu)$
	$\underline{s}(\nu) = \underline{s}(\nu-1) + \underline{v}(\nu) \cdot T_A$
Measurements at	$t = k_i T_A$, $\quad i = 1,2,3,\ldots$
KALMAN FILTER:	
Extrapolation between measurements:	$\nu = 1,2,\ldots(k_{i+1} - k_i - 1)$
	$\underline{x}(k_i+\nu) = A(k_i+\nu-1)\,\underline{x}(k_i+\nu-1)$
	$P(k_i+\nu) = A(k_i+\nu-1)P(k_i+\nu-1)A^T(k_i+\nu-1) + Q(k_i+\nu-1)$
Measurements:	$\Delta\underline{s}(k_i) = C \cdot \underline{x}(k_i) + \underline{r}(k_i)$
Extrapolation and update at measurement instances k_i:	
	$\underline{x}^*(k_i) = A(k_i-1)\,\underline{x}(k_i-1)$
	$P^*(k_i) = A(k_i-1)P(k_i-1)A^T(k_i-1) + Q(k_i-1)$
	$K(k_i) = P^*(k_i)\,C^T(C\,P^*(k_i)\,C^T + R(k_i))^{-1}$
	$\underline{x}(k_i) = \underline{x}^*(k_i) + K(k_i)(\Delta\underline{s}(k_i) - C\,\underline{x}^*(k_i))$
	$P(k_i) = (I - K(k_i)\,C)\,P^*(k_i)$
where	$A(\nu) = F(\nu T_A)\,T_A$
	$Q(\nu) = E\{\underline{n}(\nu)\,\underline{n}^T(\nu)\}$
	$R(k_i) = E\{\underline{r}(k_i)\,\underline{r}^T(k_i)\}$

Table 2: Doppler nav. and Kalman Filter algorithm

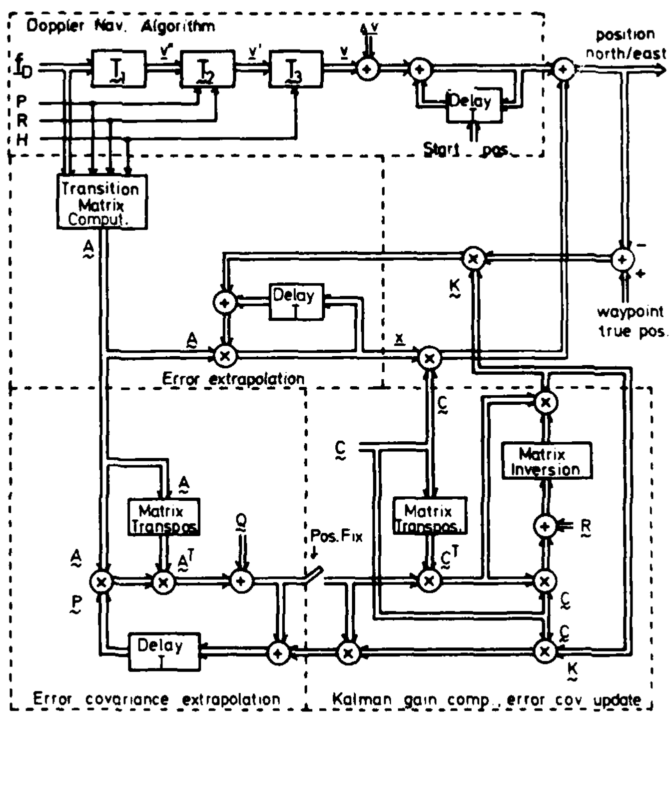


Figure 11: Information flow diagram, Doppler nav. and Kalman Filter

The algorithm has been implemented and tested off line by means of recorded flight data. An example is shown in Figs. 12 and 13. In Fig. 12, the nominal flight path is depicted as straight lines between waypoints together with the flight path as indicated by the Doppler navigation system. The difference between the nominal and indicated position, however, is meaningful only at the waypoints, where the actual positions are the waypoints themselves, whereas the indicated positions include the error of the Doppler navigation system. In Fig. 13, the absolute position errors are depicted versus the flight time, assuming a linearly increasing error behaviour.

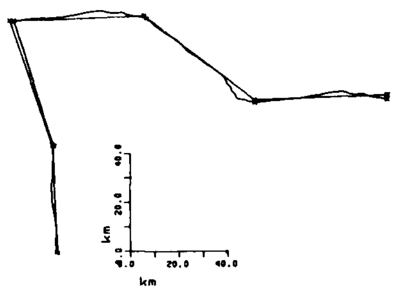


Figure 12: Flight path example composed of individual flight legs
Nominal an indicated flight path

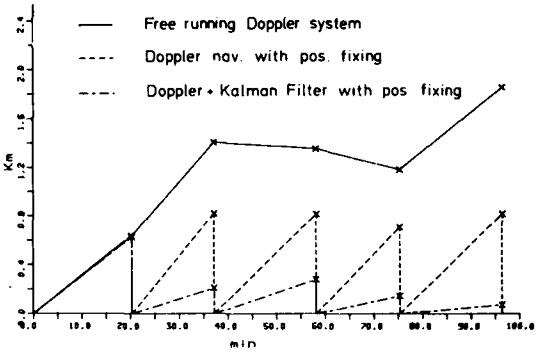


Figure 13: Position errors versus time for flight depicted in fig. 12

From Fig. 13 it can be seen that a considerable improved accuracy is achieved with a Doppler navigator that is extended by a Kalman Filter to integrate position fix information as compared to an unsupported Doppler navigator.

In terms of circular error propability values (50 % CEP), the resulting position errors averaged over all processed test flight data have been found to be 1.4 % for the unsupported Doppler navigator and 0.43 % for the Doppler navigator extended by the Kalman Filter. This result demonstrates that stand alone Doppler navigation as is in use today provides a considerable potential for improving the accuracy without any extra navigation sensors.

SUMMARY AND CONCLUSION

Based on the common practice of position fixing which is used in conjunction with Doppler navigation, a signal processing method has been presented that provides improved navigation accuracy without more accurate or any additional sensors. The occasionally available position fix information is integrated into the Doppler navigator by means of a Kalman Filter that estimates and compensates the Doppler navigation errors. Loosely speaking, a kind of inflight calibration is performed by this approach.

The achieved navigation accuracy meets a "less than 1 % CEP" requirement, so that this improved stand alone Doppler navigator can compete with more expensive approaches involving for instance inertial sensors, as long as no further requirements are imposed.

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STANDARD ATTITUDE HEADING REFERENCE SYSTEM (SAHRS) FULL SCALE DEVELOPMENT PROGRAM

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SUMMARY

There is a recognized need within the military services for reliable, low cost-of-ownership Attitude Heading Reference Systems (AHRS) capable of operating for extended periods without the need for calibration or regularly scheduled maintenance. In recognition of this need, the military services have embarked upon a joint service full scale engineering development program to provide a Standard Attitude Heading Reference System (SAHRS) utilizing strapdown technology for a multiplicity of rotary and fixed wing platforms. It is the objective of this paper to describe the system design concept, performance characteristics, and installation approach for placing SAHRS in selected military aircraft. It also dicusses the program organization, procurement approach, and schedules leading toward fully qualified military hardware.

INTRODUCTION

The AHRS is the primary navigation aid on most helicopters and it is used as either a primary aid or backup for the Inertial Navigation System (INS) on many fixed wing aircraft. Historically, AHRS in the government inventory have been developed for unique applications with little or no consideration for commonality. Although functionally similar, these devices are logistically unique. The majority of AHRS in use today are electro-mechanical-analog devices which exhibit low reliablity and are difficult and costly to maintain, both from the standpoint of obtaining spare parts as well as maintaining production lines to support the outdated technology. Consequently, a number of AHRS in all the services are approaching the end of their service life and are being considered for retrofit. Recent technical and budget actions by the services indicate additional unique AHRS are being proposed to satisfy their future needs. This combination of activities presents an "opportunity window" for acquiring a SAHRS that can be used for retrofit applications as well as for use on future aircraft.

This paper presents a cost effective approach for the development of a new generation SAHRS that can replace existing multi-box AHRS Weapon Replaceable Assemblies (WRA's) with a Strapdown Sensor Assembly (SSA) and an optional Interface Adapter Box (IAB), without requiring any changes to existing aircraft wiring. It is believed that the design approach allows for the development of an affordable system that meets the retrofit requirements of today's aircraft, and the forward fit requirements of tomorrow's aircraft.

PROGRAM NEED

Many of the front-line military aircraft have been in service for more than 20 years - far longer than was anticipated when they were first introduced. Today, the three services are faced with the challenge of extending the life of these aircraft and their associated avionic systems well into the 1990's. Avionics obsolescence and spare parts availability are now beginning to adversely affect the operational readiness of these aircaft. In the Navy, for example, the overwhelming majority of AHRS currently in use are based on late 1950 and 1960's technology. They have relatively low Mean Time Between Failure (MTBF) rates and are subject to the deleterious effects of vibration and shock.

Figure (1) illustrates the reliability problems on three of the most widely used AHRS in the Navy inventory; the AN/ASN-50, A/A24G-39, and AN/ASN-107. A sampling of the aircraft using these sytems is also shown. Note that the reliability, expressed in terms of MTBF, is not only failing to get better, but is in fact getting worse with time. The AN/ASN-50, with primary application in rotary wing aircraft, has been out of production for several years although spare Weapon Relaceable Assemblies (WRA's) and Shop Replaceable Assemblies (SRA's) are still available from the manufacturer. The AN/ASN-107 is also out of production and will require several expensive design changes to keep the system operational. The A/A24G-39 AHRS, while still in production, is not an all attitude AHRS and i unsuitable for some applications. Other negative characteristics include the use of an extensive number of WRA's, lengthy calibration procedures, accuracies which are less than those achievable with today's technology, lack of a standard digital interface capability, and inadequate analog interface characteristics.

PROGRAM BACKGROUND

For several years the military services have explored the concept of a standard AHRS for both retrofit and forward fit aircraft applications. In 1981 the Army initiated a flight test program evaluating prototype strapdown attitude heading reference systems from Litton Industries, Singer-Kearfott, Lear Siegler, and Sperry (reference a). Similarly, the Navy conducted a successful flight test program on a CH-53 helicopter in 1981 (reference b). These tests proved the feasibility of using strapdown technology for AHRS applications in helicopter missions. The Army, in fact, has introduced the Litton LR-80 AHRS in their AH-1, UH-60, and AH-64 programs.

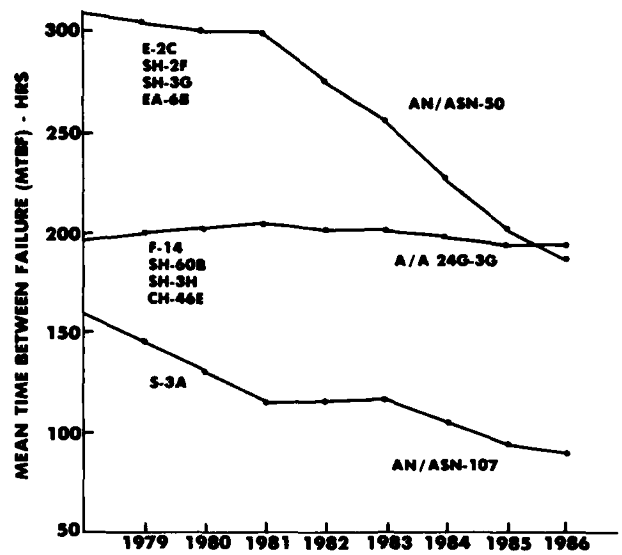


Figure 1. Attitude Heading Reference System Reliability

Subsequent life cycle cost analyses performed by the Naval Air Development Center have shown that a government developed SAHRS has significant cost savings over unique AHRS developed independently by several contractors (reference c). In the referenced study, a cost comparison was made between the two development approaches on the basis of total Research Development Test and Evaluation (RDT&E), production (non-recurring and recurring), and operating/support costs. The study assumed fixed unit production costs for a buy of 1000 systems procured over a 20 year period. The reuslts of this study are shown in Figure 2. The initial point and lower slope exhibited by SAHRS for the period 1984-1986 reflect the lower RDT&E costs associated with the single source development approach. Lower production and support costs including learning curve theory, inventory management, and support equipment costs account for the lower slope exhibited by SAHRS between 1986 and 1992. After 1992 the respective curves increase at a constant slope. This reflects the assumption that all systems exhibit similar reliability characteristics and that spares replenishment schedules for both the proposed standard SAHRS and the unique AHRS are identical. As shown in the figure the resulting total life cycle cost difference over 20 years is approximately $100 million less for the standard AHRS approach.

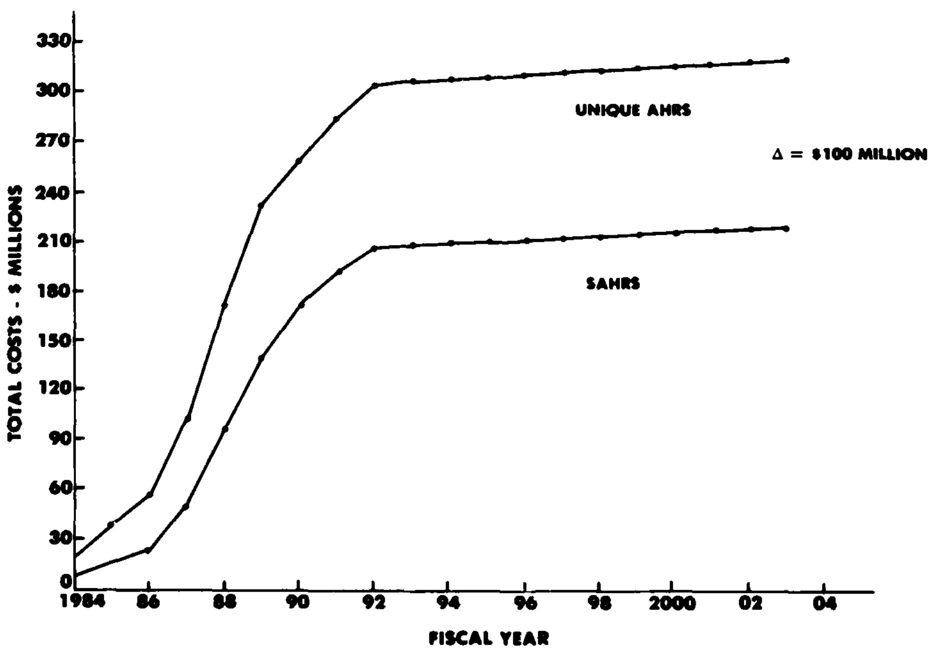


Figure 2. Life Cycle Cost Comparison

In 1982 the Naval Air Systems Command tasked the Naval Air Development Center to initiate a study effort investigating the feasibility, practicality, and retrofit requirements for replacing existing AHRS with a new SAHRS on selected helicopter and fixed wing aircraft (reference c). Helicopters selected for this study were the SH-2F, SH-3H, CH-46E, and RH-53D. The fixed wing aircraft were the S-3A and C-130G; later, the SH-60B, F-14A, and E-2C aircraft were added to the study. The results of this study (reference d) formed the basis of the design approach taken on the SAHRS program.

PROGRAM CHARTER

The program was formally endorsed by the Standardization Panel for Avionics at the January 1983 planning conference of the Joint Services Review Committee (JSRC) on Avionics Components and Subsystems (AVCS). Responsibility for program management and execution was assigned to the U.S. Navy at that time. During 1983 technical teams from each of the three services, working together, drafted a combined military specification, MIL-R-85632 (AS), stating the requirements for the system. Separate appendices were attached to this specification describing the peculiar interface requirements for each of the aircraft initially selected to receive SAHRS. The designated Navy aircraft were the SH-2F, SH-60B, F-14A, and E-2C, and for the Army it was the UH-1 helicopter.

The Air Standardization Coordinating Committee, made up of representatives from the United States, Australia, Canada, and New Zealand, are proposing the specification as a standard within their countries. They are also planning to forward it for consideration as a standard to the appropriate technical working groups of both the North Atlantic Treaty Organization (NATO) and South East Asia Treaty Organization (SEATO) member nations.

PROGRAM OBJECTIVES

The SAHRS program has been initiated to provide the military services with a common, price competitive, reference system for use on multiple platforms. The program will take advantage of recent technological developments in strapdown sensor technology that hopefully will result in improved equipment readiness and reduced life cycle costs. Specifically, the program is intended to accomplish the following objectives:

- Design, develop, test and obtain Approval For Production (AFP) of a new SAHRS, type designation AN/USN-2.

- Develop a standardized interface that will be compatible with existing applications as well as meeting future aircraft needs through incorporation of a modular Input/Output section containing analog, discrete, and MIL-STD-1553B digital interfaces.

- Provide a system with increased reliability (2000 hrs MTBF) and with no scheduled maintenance or calibration requirement.

- Provide the system at low cost ($40,000 or less in quantities of 1000 or more) and with growth capability to a moderate accuracy inertial navigation system.

PROGRAM ORGANIZATION

The executive service in charge of managing the program is the U.S. Navy with the Army and Air Force participating as deputy program managers. Figure 3 illustrates the top level organization and staffing. Overall program management authority resides with the Program Manager. He is repsonsible for delegating specific tasks related to overall program execution. The AVCS Program Manager is repsonsible for coordinating project activities (status, funding, schedules, etc) with the Joint Services Review Committee, which oversees this program as well as other joint service programs under their jurisdiction. The Deputy Program Manager for Acquisition is in charge of the day-to-day execution of the project. He is required to have a comprehensive understanding of the needs and requirements of each of the services involved. He must also understand the differences in areas such as financial management, program management philosophy, logistics support, organization, and test and evaluation techniques so that he can arbitrate and resolve any potential problems as they may arise. The Deputy Progam Manager for each service has broad systems engineering repsonsibility insuring that the developed equipment meets the individual service needs. The Deputy Program Managers are in turn supported by system engineers from their own organizations that are knowledgable in specific hardware/software disciplines.

A key systems engine function is provided by the Interface Coordination Working Group (ICWG). This group is cha by the Navy and is composed of representatives from the three services, airframe manufacturers, and contractor(s) personnel. It is their responsibility to generate an Interface Control Document (ICD) defining the required interface between SAHRS and the associated avionics of each selected aircraft.

Due to current funding uncertainties within the Army and Air Force, the primary Research Development Test and Evaluation (RDT&E) costs and initial production costs are being provided by the Navy. The program will continue while the other two services evaluate their system needs and funding priorities.

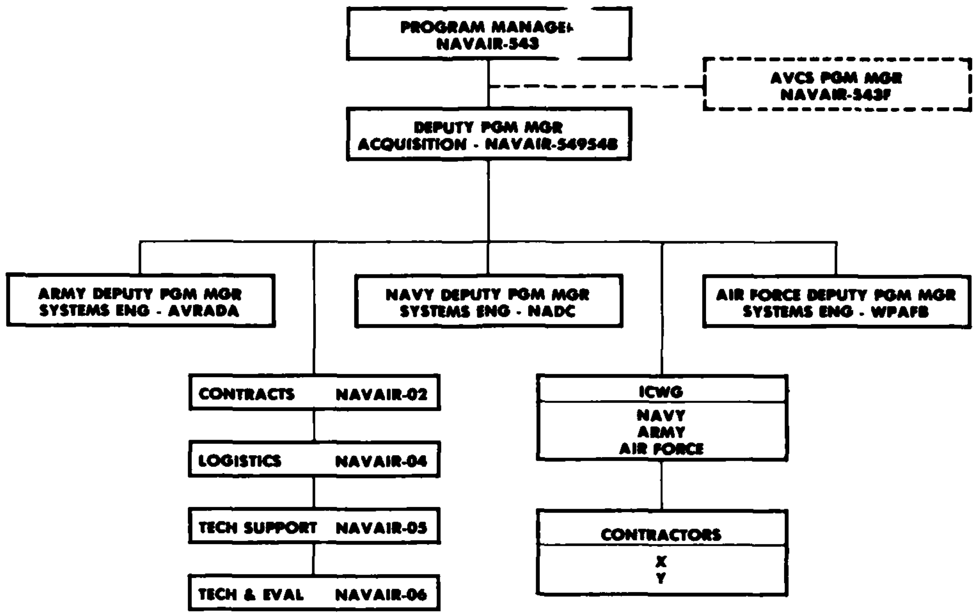


Figure 3. SAHRS Program Organization

PROCUREMENT APPROACH

A two phased procurement approach is planned: Phase I - Preproduction and Phase II - Production. In the Preproduction or Full Scale Development phase, two contractors, chosen by competitive proposal evaluation, will develop, fabricate, and conduct design approval testing on their respective systems. Multiplex data bus qualification testing will be conducted by a government laboratory equipped to thoroughly exercise this equipment capability. All other testing will be conducted at the contractor/government test facilities as determined by the Test Program Manager. Also during this phase the contractor will perform logistic support analyses to better define alternate logistic support approaches for implementation during the Production phase. Phase I will culminate with aircraft integration, compatibility and flight tests conducted as part of the Technical Evaluation (TECHEVAL) and Operation Evaluation (OPEVAL) qualification procedure. Upon successful completion of these tests the systems will then receive formal Approval for Production (AFP).

Phase II will be a competitive production acquisition among the two qualified contractors (assuming both successfully pass Phase I). Requests for production proposals will go out near the end of TECHEVAL so that the production contract can be awarded shortly after OPEVAL. It is anticipated that the initial production contract awards will be split in a manner commensurate with respective bid prices. Every year as production options are exercised, each manufacturer or any other qualified manufacturer can openly bid with contract awards going to the lowest bidder.

GIMBALLED VERSUS STRAPDOWN AHRS - A CONCEPTUAL COMPARISON

In order to fully understand the SAHRS design it is useful to first compare the major differences between gimballed and strapdown AHRS. The theory of operation for both is given below.

A. Gimballed AHRS Concept

A conceptual sketch of a gimballed AHRS similar to the AN/ASN-50 is shown in Figure 4. The system utilizes a vertically oriented Vertical Gyro (VG) to sense aircraft roll and pitch, while aircraft heading is sensed by the horizontally oriented Directional Gyro (DG). A system of mechanically gimballed platform torquer motors and associated bubble level sensors mounted on gimbal arm maintain the VG and DG at the proper orientations. Aircraft attitudes are provided by the angular resolvers which measure the angular difference between the airframe and the stable gyro platform.

During normal non-maneuvering flight, the VG is prevented from drifting away from the vertical orientation via the "erection" loops. For example, a platform tilt in the pitch direction is sensed by the bubble level switch mounted on the pitch gimbal. The error signal is transmitted to the pitch erection loop amplifier which then commands the pitch erection torquer to rotate the roll gimbal, thereby causing the VG to precess back to the vertical orientation. Similarly, the DG is normally precessed to the magnetic heading as measured by the flux valve via the "slaving" torquer circuit (see Figure 4).

During periods of acceleration and maneuvers, the pendulous design of the flux valve causes false magnetic heading outputs and thereby generates erroneous slaving signals to the DG. In addition, the bubble level erection switches will sense apparent non-verticality due to maneuvers and generate improper erection signals to the VG. Thus, it is necessary to sense these periods

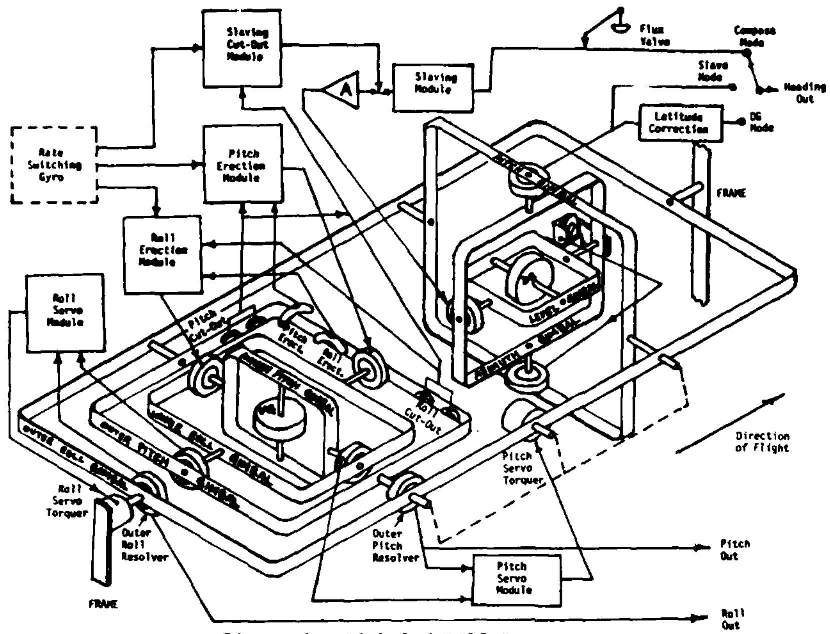


Figure 4. Gimbaled AHRS Concept

of non-benign flight conditions and cut out the erection and slaving signals to the VG and the DG, repsectively. This is accomplished with either bubble level cut-out switches and/or separate rate switching gyro as shown in Figure 4.

There are three primary AHRS modes of operation defined by the status of the DG as follows:

1. Slaved Mode
2. Free or Directional Gyro (DG) Mode
3. Compass Mode

The Slaved mode is the normal mode of operation. In this mode, the Directional Gyro (DG) azimuth signal is combined with the flux valve output to produce an integrated magnetic heading. Since the DG has good short term stability while the flux valve is consistent over the long term, the slaving operation produces the best heading accuracy. During high latitude (>70°) operations and/or aircraft maneuvers, the Free or DG mode of operation becomes necessary. In this mode, the output of the DG is used directly to provide heading information. However, since the magnetic north reference is no longer available, there is an apparent heading drift due to the earth's rotation. The drift rate varies as a function of latitude and a latitude correction sig-nal is supplied to minimize the apparent heading error. Finally, during periods of DG failure, the compass mode of operation is entered. In this mode, the flux valve output is used directly to generate the heading information.

B. Strapdown AHRS Concept

Advances in microprocessor and high dynamics gyro technology have made possible the implementation of the Strapdown AHRS concept as shown in Figure 5. In the strapdown mechanization, the gyros and accelerometers are tied directly to the airframe and mechanical gimbals are replaced by a strapdown processor which mathematically maintains the sensor assembly in a local level orientation. The Strapdown AHRS is a Schuler-tuned, second order system similar to an inertial navigation system mechanization. The system senses vehicle angular rates and translational accelerations by means of two - 2 degree of freedom dry-tuned gyroscopes or three single axis laser gyroscopes, and three single axis force rebalanced accelerometers. Aircraft attitude is computed via continous integration of attitude rate data from the gyros while being compensated for aircraft accelerations that are accurately measured by the accelerometers. The aircraft attitude is computed in the form of a direction cosine matrix relating body to navigation coordinate axes.

The initial attitude of the SAHRS is determined via the gyro-compassing alignment procedure similar to that used in inertial navigation systems. In this procedure, the SAHRS processor uses initial latitude data to compute the components of earth's rate, which along with components of gravity sensed by the accelerometers, are used to derive the initial roll, pitch, and true heading of the SAHRS.

Since the SAHRS will be mechanized similar to an inertial navigator with computations performed within its own processor, it is a straightforward process to upgrade the system into a moderate quality INS. System performance can then be enhanced by combining the inertially dervied velocity with doppler velocity in a hybrid SAHRS/doppler configuration.

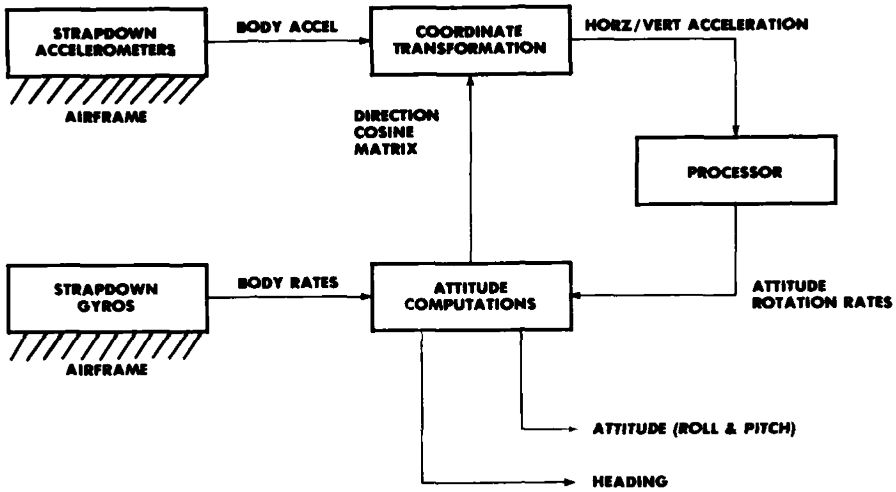


Figure 5. Strapdown AHRS Concept

SYSTEM DESCRIPTION

The SAHRS provides analog and digital outputs of aircraft pitch, roll, heading, angular rate. angular acceleration, linear acceleration, velocity, and position. It will have the capability of providing inertially smoothed velocity data when used in conjunction with a velocity reference such as a doppler radar set. The SAHRS will also be capable of slaving inertial heading to a Magnetic Azimuth Detector or flux valve input. In production, the sytem will consist of up to four Weapon Replaceable Assemblies (WRA's) depending on the selected aircraft configuration. Figure 6 illustrates the four box configuration consisting of the following:

a. Strapdown Sensor Assembly (SSA) - Standard

b. Mode Control Unit (MCU) - Optional

c. Mount - Optional

d. Interface Adapter (IA) - Optional

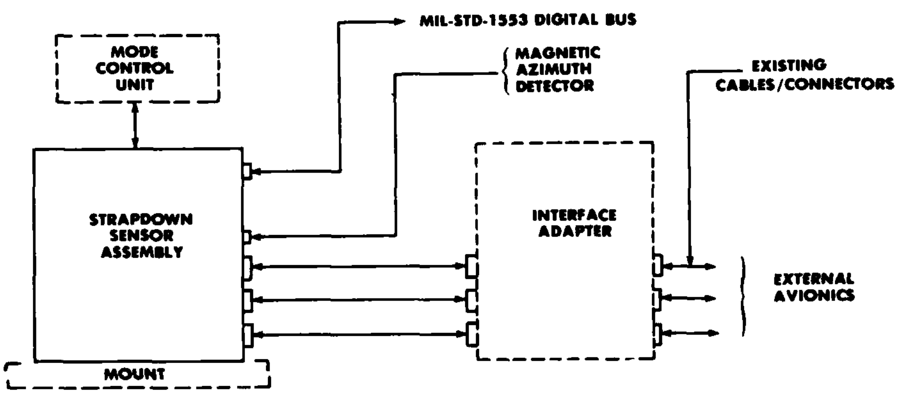


Figure 6. Full SAHRS Configuration

A. Strapdown Sensor Assembly (SSA)

Figure 7 depicts the functional diagram of the SSA which consists of two sections; the core section and the interchangeable section. The core section is common to all SAHRS and contains the sensor package, power supply, control electronics, processor, and Input/Output interface. The aircraft body rates are measured by the sensor assemblies and transmitted to the control electronics where sensor signal conditioning functions take place. Depending on the contractor(s) design approach, the sensor package may contain two 2-degree of freedom dry-tuned gyros or three single axis laser gyros. Three single axis force rebalanced accelerometers will be provided in lieu of bubble levels used in conventional gimballed AHRS. The power supply provides basic power to the sensor package, control electronics and other cards as required. The

control electronics also transmit the digitized sensor data to the processor where body rate information is integrated and processed via software to produce aircraft attiutdes, rate, and heading information. These data are then transmitted to the interchangeable section via the generalized processor I/O interface. The interface formats all I/O data and interacts with the digital bus. The digital bus wiring is identical to all cards and can be implemented using the unibus concept incorporated by modern minicomputers.

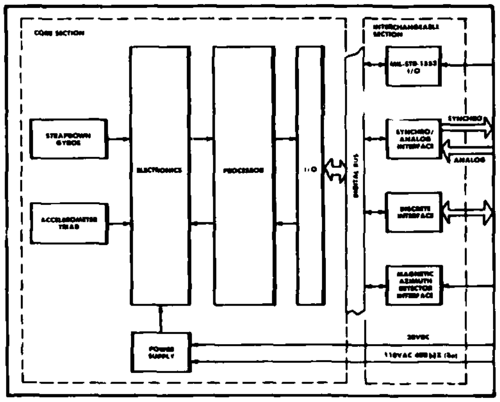


Figure 7. Functional Diagram of Strapdown Sensor Assembly

The Interchangeable Section is made up of interchangeable cards tailored to the particular vehicle and transparent to the digital bus. It is also designed to accept new cards/functions during avionics upgrade programs without requiring any internal wiring changes, software changes or reprogramming. The interchangeable section provides the following functions as required: a MIL-STD-1553 multiplex data bus, a synchro/ analog interface, dicsretes, and magnetic azimuth detector interface.

B. Mode Control Unit (MCU)

This unit provides the controls and indicators required for operating SAHRS. Use of the MCU is optional. It is planned only for those aircraft desiring full, self-contained SAHRS capability. In new aircraft, selection of the SAHRS modes may be accomplished by an integrated control panel via the MIL-STD-1553B mux bus in lieu of a self-contained MCU. In retrofit aircraft, where it is desireable that SAHRS operation be indistinguishable from the current AHRS, use of the existing control unit is planned. Some limitations will occur, however, which could inhibit full realization of SAHRS potential when used in this manner. For example, gyrocompassing alignment and stored heading alignment cannot be initiated as these modes are not available on existing controllers.

C. Mount

The Strapdown Sensor Assembly (SSA) is hard mounted to the aircraft through the optional mount or on a stand alone basis. The mount will be designed to allow installation of the SSA with an alignment repeatability of 3 arc minutes (3 sigma) in any axis. It will also have a quick disconnect provision to minimize removal and installation time.

D. Interface Adapter (IA)

The Interface adapter functions as an optional signal conditioning/ junction unit for interfacing the SSA with external avionics equipment. It will be designed to eliminate the need for rewiring existing aircraft harnesses and connectors in retrofit applications. U.S. Navy aircraft initially scheduled to use the IA are the SH-2F, E-2C, and F-14A. The IA will contain interface cards and/or signal amplifiers for driving the required avionics equipment such as heading and attitude indicators. As a minimum the IA will accept roll, pitch, and heading information from the SSA and output these signals in the required format for each aircraft. It is expected that the cost of the IA will be low compared to the SSA and MCU. Thus during future MIL-STD-1553B multiplex bus upgrades of the aircraft, F-14A to F-14D for example, the IA can be "thrown away" and SAHRS will then interface directly with the multiplex data bus.

DYNAMIC OPERATING RANGES

The dynamic operating ranges specified for SAHRS are shown in Table 1. The velocity range is sufficiently broad since both helicopters and fixed wing aircraft are using the system. Roll information will be continuous throughout plus or minus 180° and valid for all vehicle angles of pitch and heading. It is noted that for roll rates over 200°/sec lasting more than three seconds, degraded performance is allowed. This condition is permitted since it is unlikely the aircraft will exceed these limits under normal flight maneuvers. Pitch information will be

continuous through plus or minus 90° and valid for all angles of roll or heading. Heading information will be continuous throughout 360° and valid for all angles or pitch and roll.

PARAMETER	MISSION CONDITION
LATITUDE RANGE (DEGREES)	S90 TO N90
LONGITUDE RANGE (DEGREES)	W180 TO E180
LONGITUDINAL VELOCITY (METERS/SECOND)	−30 TO +750
LATERAL VELOCITY (METERS/SECOND)	−750 TO +750
VERTICAL VELOCITY (METERS/SECOND)	−75 TO +75
ROLL ATTITUDE (DEGREES)	−180 TO +180
PITCH ATTITUDE (DEGREES)	−90 TO +90
YAW (DEGREES)	0 TO 360
YAW ANGULAR RATE (DEGREES/SECOND)	0 TO ±200
ROLL ANGULAR RATE (DEGREES/SECOND)	0 TO ±300*
PITCH ANGULAR RATE (DEGREES/SECOND)	−150 TO +150
PITCH ANGULAR RATE THROUGH ZENITH (DEGREES/SECOND)	−50 TO +50
ROLL ANGULAR ACCELERATION (DEGREES/SECOND^2)	0 TO 1500
PITCH AND YAW ANGULAR ACCELERATION (DEGREES/SECOND^2)	0 TO 400
LINEAR ACCELERATION RANGE (METERS/SECOND^2)	0 TO 100

*PERFORMANCE DEGRADATION FOR ROLL RATES OVER 200 DEG/SEC LASTING MORE THAN 3 SECONDS IS PERMISSIBLE

TABLE I. SAHRS DYNAMIC OPERATING RANGES

MODES OF OPERATION

The SAHRS will be capable of operating in the following operator initiated modes:

a. Gyrocompass mode. The gyrocompass mode is the primary operating mode of the system. When in this mode automatic selection of the alignment submode including gyrocompass only alignment, stored heading alignment or magnetically slaved initialized gyrocompass alignment for rapid reaction applications is made.

b. Slaved mode. The slaved mode of operation provides attitude and gyro-stabilized magnetic heading outputs. In the slaved mode, the heading gyro is continuously slaved to agree with the magnetic heading sensed by the Magnetic Azimuth Detector (MAD). The system will also be capable of rapidly synchronizing to the heading output from the MAD.

c. Directional gyro mode. When operated in this mode, the SAHRS provides earth rate corrected directional gyro heading. This mode is typically selected during high latitude (>70°) flight operations.

d. Compass/Emergency mode. The compass/emergency mode is a backup mode in the event of a failure of the inertial sensing elements. When operated in this mode, heading outputs are electronically damped from the unstabilized magnetic heading.

e. Calibration mode. This mode provides for the selection of two sub-calibration modes of operation: (1) the automatic compass swing and heading calibration of the MAD without the need for external compass calibration equipment, and (2) the heading calibration of the MAD with external compass calibration equipment. Calibration information for both sub-modes will be stored in a removable SRA of the SSA or IA. On aircraft with a MIL-STD-1553 data bus, calibration information will be made available for storage in the bus controller. A dedicated test connector will be provided to allow connection of either the Air Force MC-1 or the Navy MC-2 compass calibration equipment.

The test connection will provide the necessary interface for conducting an electrical compass swing to determine the one cycle and two cycle compensation factors.

f. In-flight alignment gyrocompass mode. The SAHRS will have an in-flight gyrocompass capability when valid position or velocity and latitude inputs are available.

g. In-flight restart with and without aid mode. The SAHRS will be capable of in-flight with restart and without aiding sensors. With valid magnetic azimuth detector and velocity or position inputs, aided alignment shall meet the in-flight requirements specified in Table II and Table III. Without aid (only manual inputs) the SAHRS shall align within two minutes from -18°C and provide the accuracies specified for directional gyro operation in Table II.

SAHRS OUTPUT ACCURACY

The SAHRS specified output accuracy for the operational modes described above is shown in Table II. It is noted that the heading accuracies specified in the gyrocompass, slaved, and in-flight restart modes assume that velocity or position aiding is continuously available. This is

normally the case for U.S. Army helicopters that rely on doppler velocity data for velocity damping computations. When aiding is not available, as is the case on most U.S. Navy helicopters and fixed wing aircraft, or it is unreliable, an additional 0.25° RMS should be added to these accuracies. Heading accuracies in the slaved mode are based upon the use of an ML-1 Magnetic Azimuth Detector (MAD) or equivalent with compass deviation compensation provided in the SAHRS. Earth's magnetic field anomalies are also excluded, and perfect alignment between the SAHRS, doppler, and MAD are assumed.

PARAMETER	SIGNAL	GYROCOMPASS	MODES (ACCURACIES IN RMS VALUES) SLAVED	INFLIGHT RESTART	DIRECTIONAL GYRO	COMPASS/ EMERGENCY
MAG/TRUE HEADING (DEG)	SYNCHRO/ DIGITAL	0.5	0.75	0.75	0.4°/HR	1°
PITCH, ROLL (DEG)	SYNCHRO/ DIGITAL	← 0.25 →				
YAW, PITCH, ROLL ANGULAR RATE (DEG/SEC)	DIGITAL	← 0.25 →				
YAW, PITCH, ROLL ANGULAR ACC. (DEG/SEC^2)	DIGITAL	← 16.0 →				
LONGITUDINAL, LATERAL/NORMAL & VERTICAL VELOCITY (METERS/SEC)	DIGITAL	← 0.2 →				
LONGITUDINAL, LATERAL, NORMAL & VERTICAL ACC (g)	DIGITAL	← ±0.2 →				

Table II. SAHRS Output Accuracy

ALIGN SUBMODE	HEADING ACCURACY (RMS)	AMBIENT TEMPERATURE −18°C	−40°C	−54°C
GYROCOMPASS	0.5°	8 MIN.	11 MIN.	13 MIN.
MAG. SLAVED INITIALIZED G.C.	0.5°	2 MIN.	3 MIN.	4 MIN.
STORED HEADING	0.5°	2 MIN.	3 MIN.	4 MIN.
SLAVED	0.75°	2 MIN.	3 MIN.	4 MIN.
IN FLIGHT RESTART				
WITH AIDING	0.75°	2 MIN.	3 MIN.	4 MIN.
WITHOUT AIDING	0.4°/HR	2 MIN.	3 MIN.	4 MIN.
DIRECTIONAL GYRO	0.4°/HR	2 MIN.	3 MIN.	4 MIN.
COMPASS/EMERGENCY	1°	0.5 MIN.	0.5 MIN.	0.5 MIN.

Table III. SAHRS Reaction Times

SAHRS REACTION TIMES

SAHRS reaction times, e.g. the total elapsed time from system turn-on to the availability of specified output data, for the modes of operation listed in Table II, are shown in Table III. It is noted that the reaction times for temperatures above -18°C to +71°C are the same as the -18°C requirements. Further, the heading accuracy for the Slaved mode and In-Flight Restart with aiding assumes continuously available magnetic heading and velocity or position information. In-flight restart without velocity or position aiding also assumes perfect magnetic variation and magnetic heading manual entry inputs.

BUILT-IN TEST

The SAHRS will feature a comprehensive Built-In-Test (BIT) capability to provide both automatic and operator initiated system test and fault isolation without the use of external test equipment. In addition it will provide the capability to assess the functional performance and

to identify failed modes in order to determine operational readiness or the ability to complete the aircraft mission. To permit assessment of a detected failure, a discrete GO/ NO-GO signal and a digital representation of the failed mode will be displayed on the SSA. This information will also be made available as an output for presentation on the operator's Mode Control Unit (MCU) or MIL-STD-1553B combined display.

There are three distinct BIT modes of operation included in the design: Initial BIT, Periodic BIT, and Initiated BIT. A brief description of these modes follows:

a) Initial BIT - SAHRS will sequence through an automatic BIT at turn-on to establish functional integrity. This routine emphasizes the testing of components/modules that can cause total loss of SAHRS capability. Sufficient test provisions will be included so that at least 95% of all electronic system failures result in a NO-GO indication with a 2% false alarm rate.

b) Periodic BIT - Periodic BIT operates automatically after completion of Initial BIT checking for operability every five (5) seconds. At least 95% of all system failures will result in a NO-GO indication with a 2% false alarm rate.

c) Initiated BIT - The initiated BIT routine is in effect a first level maintenance activity initiated by the operator on the ground. In this mode normal system operation may be interrupted to complete the BIT check. At least 98% of all equipment failures shall be detected and isolated to the Shop Replaceable Assembly (SRA) e.g. card level with a 1% false alarm rate.

RELIABILITY

Improved reliability is one of the principal objectives of the SAHRS program. As mentioned previously the low reliability exhibited by current AHRS (200 hours MTBF or less) is having an adverse effect on the operational readiness of many Navy aircraft. In recognition of and in response to this deficiency, a major reliability development plan will be implemented in accordance with MIL-STD-785 (Reliability Program for Development and Production Systems) that has as its goal a specified MTBF ($_0$) of 2000 hours and a minimum acceptable MTBF ($_1$) of 1000 hours.

The program plan and implementation thereof will be periodically reviewed, updated, and revised as required throughout the Full Scale Development program. In addition, a Failure Review Board will be established to review data on all failures occurring on preproduction systems to insure that corrective action is taken prior to beginning production. The following major reliability program elements will be included under the contract:

1. Reliability program plan describing the contractor's plans for conducting the program including organizational responsibilities and schedules.

2. Reliability prediction, allocation, and analyses including stress, parts/circuit tolerance analysis, e.g. worst case analysis, and sneak circuit analysis.

3. Failure Mode and Effects Criticality Analysis (FMECA).

4. Parts control program including screening, inspection, parts derating criteria, and non-standard parts identification.

5. Power supply reliability program as a supplement to the general reliability program including elements of 2, 3, and 4 above.

6. Periodic reliability scheduled during major design reviews.

7. Environmental qualification tests based upon simulated mission pro-files, with temperature cycling, random vibration, and electromagnetic compatibility test in accordance with MIL-STD 810C requirements.

8. A closed loop data collection and analysis system to determine the failure cause and required corrective action.

It is noted that particular attention is given to Power Supply reliability because low reliability in these units has been a recurring and nagging problem not only on AHRS but also on inertial navigation systems and other communication/navigation systems as well. Further, to insure that the inherent reliability designed into the equipment is maintained, a warranty program is being considered for the production phase. The warranty requirements are being identified at this time and will be included in the pilot production contract.

MAINTAINABILITY

The maintainability objective is to design SAHRS so that it may be kept operational or restored to operational status with a minimum of maintenance resources. A firm program requirement is that SAHRS be designed for ease of maintenance and that it not require scheduled recalibration or maintenance. Modular construction will be employed wherever possible to facilitate repair by replacement of cards or modules. Sensor assemblies and cards will be mounted such that replacing one does not require removal of the others. It will also be designed such that all WRA's are replaceable at the organizational level and that boresighting will not be required after replacement of the WRA. Maintenance characteristics including maintenance times specified for the organizational and intermediate level are shown in Table IV.

	ORGANIZATIONAL LEVEL	INTERMEDIATE LEVEL
MEAN TIME TO REPAIR (MTTR) 90% CONFIDENCE LEVEL	0.25 HRS	0.75 HRS
MAX CORRECTIVE MAINTENANCE TIME 90% CONFIDENCE LEVEL	0.50 HRS	1.5 HRS
OPERATIONAL SERVICE LIFE	NOT LESS THAN 10,000 HRS	

Table IV. SAHRS Maintenance Characteristics

The maintenance times shown above do not include the time to actually repair an SRA, e.g. replace a discrete component on a card. Operational service life is defined as the total operating time between the start of operation and wear out of the equipment. Wear out is defined as the point where overhaul or repair costs exceed one-half of the replacement costs of the equipment.

The present maintenance concept, pending reliability and unit cost, is to utilize the so called conventional or three level organic maintenance approach implemented as follows:

a) Organizational Level - Remove and replace faulty WRA's using Built-In-Test. System GO/NO-GO shall include WRA interface verification by means of the self-test diagnostic software program.

b) Intermediate Level - Repair WRA's by removal and replacement of faulty SRA's using Built-In-Test capability.

c) Depot Level - Repair of WRA's/SRA's beyond the intermediate level at a government rework facility.

INTEGRATED LOGISTICS SUPPORT (ILS)

ILS planning requirements are being established by a Navy logistics group with application to the tri-service program. A detailed specification has been generated establishing ILS requirements necessary to support the program through TECHEVAL/OPEVAL. It establishes the methodologies and procedures by the contractor shall use in the implementation of ILS, and includes documentation required to meet the specific needs of the Navy, Army, and Air Force.

During the full scale development, a Logistics Support Analysis (LSA) will be conducted by the contractor(s) to define a recommended logistic support approach for the production phase of the program. For example, a five year Reliability Improvement Warranty (RIW) will be considered as an alternate to the traditional contractor supported depot level maintenance approach. Additionally, a Level of Repair Analysis (LORA) will be conducted to determine the most effective and economical maintenance levels for the system.

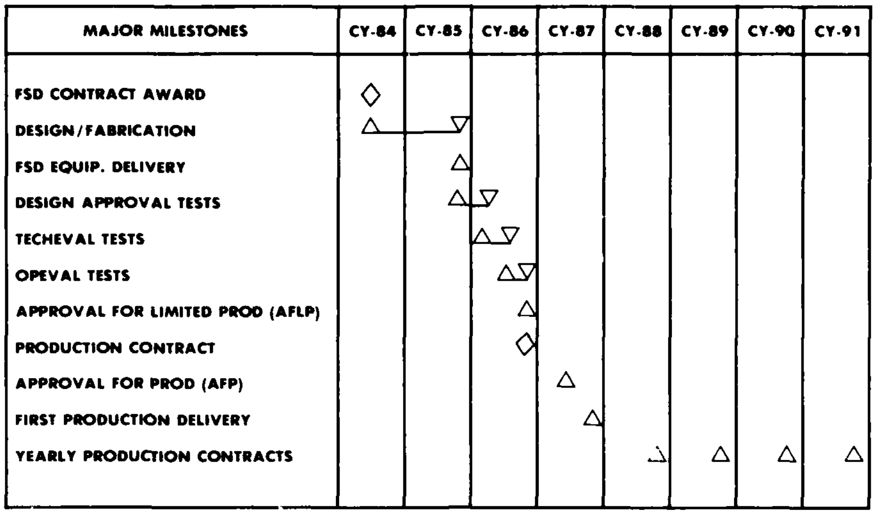
SCHEDULE

Figure 8 summarizes the major schedule milestones. Assuming contract award in July of 1984, FSD model deliveries are planned to occur between November 1985 and March 1986.

It is anticipated that an Approval For Limited Production will be obtained in Oct 1986 in order to award the first Production Contract early in Fiscal Year 1987. Initial production deliveries are scheduled in early FY-88.

COST CONSIDERATIONS

Preliminary cost analyses have been conducted using the U.S. Air Force Step II Life Cycle Cost (LCC) model that indicate an average price of $40,000 in quantities of 1000 or more is a realistic cost goal. If the capability of the system is upgraded in the future to a moderate quality inertial navigator, then an average price of $60,000 in similar quantities has been estimated. The cost risk is considered low since several manufacturers are already bidding similar equipments in this price range. As new and more accurate cost information becomes available it will be added to the model to refine the above estimates.



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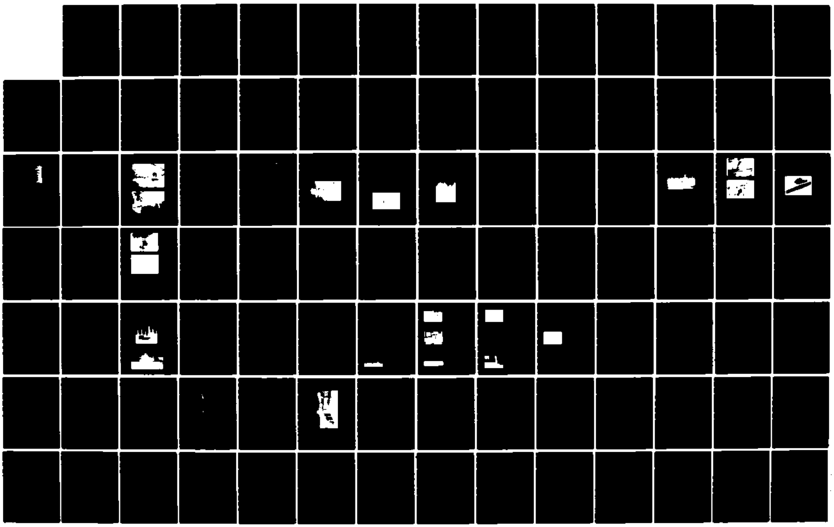

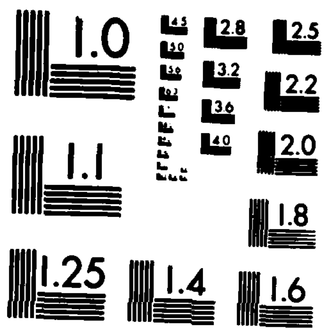

MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

INTEGRATED CONTROL AND DISPLAY SYSTEMS

FOR

HELICOPTER BATTLEFIELD MISSION MANAGEMENT

BY

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Summary

This paper discusses the avionics requirements of an integrated crew station for battlefield helicopters, with the need to manage the sub-systems providing the mission equipment package, the communications, navigation and engine and airframe monitoring. An example of the approach taken to equip a multi-role light attack helicopter is provided.

Introduction

In the cockpit of most of todays battlefield helicopters the control and display aspects of the man-machine interface are not optimised, with high levels of crew workload; technology now exists to improve the situation.

With the evolution of complex avionic architectures and the proliferation of sensors to meet today's exacting battlefield environment, an avionics fit can now account for a major proportion of the cost of a battlefield support helicopter.

To achieve mission success, an increasing number of sensors are necessary to supplement the crews' abilities and to provide the battlefield helicopter with a day and night, good and bad weather, operational capability.

Therefore, in order to maximise the effectiveness of such a helicopter, it is essential that the various avionic sub-systems and sensors should function in harmony with engine, airframe and, not least, the crew.

Thus a total systems approach is necessary for the control and display integration of the new generation of battlefield helicopters in order to meet the increasingly severe operational requirement.

For existing in-service helicopters, the retrofit of integrated avionics systems to manage much of the existing avionics will permit an extension in the useful life of the helicopter and allow new sensors and systems to be added without major redesign of the cockpit area; the availability of an avionics management system reducing the need for controllers in the cockpit.

Racal Avionics are currently collaborating with several helicopter manufacturers and helicopter operators in the development of Integrated Control and Display Systems for battlefield mission management. System configurations for new helicopters and avionic retrofits are based on the family of Racal Avionics Management Systems (RAMS).

This paper describes various aspects of RAMS when configured for the integrated crew station of a new multi-role light helicopter which is being flight tested this year. Aircrew and engineers have had the opportunity to discuss the requirements, to take into account the availability of equipment to meet the present needs and allow for the fact that future expansion and enhancement of the system will be necessary.

Requirements of an Integrated Control and Display System (ICDS)

Operational experience has highlighted the demands of maintaining communication and accomplishing accurate navigation as well as performing the assigned mission of the helicopter.

Avionics management systems can help to meet these demands and provide significant improvements by optimising the control and display of systems fitted to the battlefield helicopter.

The number of sub-system controllers is reduced and the display of data on multi-function displays increases with the integration of the avionics, all aimed at making the ICDS "user friendly", a much abused phrase but an appropriate expression when understanding the control and display environment of a cockpit.

As most of the new displays are using cathode ray tube (CRT) technology it will be necessary for aircrews to overcome the "CRT shock" of an "all glass cockpit" and come to terms with new operating procedures. The present scan pattern of a pilot when using conventional instruments must be adapted to accept the new display concepts.

As ICDS configurations enter service, it will be necessary for new training concepts to assist in the learning of new procedures; Mission Equipment Package training aids will also be necessary to assist in the familiarisation of aircrews with integrated systems.

Once aircrew are familiar with the new operating concepts, they will rapidly be able to obtain the advantages of an integrated crew station.

Typically a RAMS Integrated Control and Display System (ICDS) with MIL-STD-1553B interfacing comprises the following units,

Multi-function Display (MFD)
Control and Display Unit (CDU)
Processor Interface Unit (PIU)
Symbol Generator (SG)
Data Transfer Device (DTD) and Receptacle

An ICDS must address the following 4 major areas of required system integration,

Mission Equipment Package

Communication

Navigation

Engine and airframe monitoring

The avionics management aspects which RAMS provides for these topics is now described in more detail.

Mission Equipment Package sub-system management

On the MFD, the primary flight instrument in the cockpit, the crew is provided with displays from video sensors such as Mast Mounted Sight (MMS) and Forward Looking Infra Red (FLIR) systems. Flight symbology for attitude, heading, airspeed, altitude, torque etc. can be displayed alone or overlaid on the video from the sensors.

The co-pilot gunner, with a separate heads down display, can fire the weapon systems; the pilot and co-pilot gunner can select their source of video independently and either crew member can switch between the two sources of video to assist in the determination of possible targets.

Symbology for a horizontal situation display of the battlefield situation can be selected with one of the fixed function keys along the bottom of the multi-function display.

As well as visual sensors, data from defensive aids such as radar and laser warning receivers can be integrated onto the displays, as well as the status of the weapons carried.

Communications sub-system management

For a battlefield helicopter both short range and long range communication is required.

The ICDS enables radio mode control and frequency tuning to be centrally performed, enabling the individual radio set controllers to be removed from the cockpit.

To reduce to a minimum the necessity of removing the hands from the flying controls to change communication frequencies, modes and selection of the various radios, an ICDS provides,

HF, VHF and UHF radio frequency storage and selection
Radio mode storage and selection

Radio transmitter selection
IFF Codes storage and selection
Data Link Management

For example, a list of frequencies entered into the data base stored in the processor memory allows changes in transmission frequency to be made up or down the pre-set list using a 3 position switch, centre off, change frequency up or down.

The selected frequency or channel number is displayed on the *multi-function display or a separate remote indicator, scrolling of the display allows the required frequency to be selected.

Similarly with IFF equipment, the code changes required at the appropriate times are accomplished automatically, the codes being entered into the processor data base before flight.

Navigation sub-system management

The optimum navigation system makes the best use of autonomous (self-contained) and terrestrially referenced navigation sensors.

Accurate autonomous navigation is achieved by the integration of a light weight Racal Avionics Doppler Velocity Sensor (DVS) and a strapdown Attitude and Heading Reference System (AHRS) interacting via a full Kalman filter mix to sense helicopter attitude, heading, and the component velocities of the flight path vector.

The advantages of combining two velocity sensors with complementary error characteristics and continuous airborne gyro compassing of the AHRS thus bounds both heading and attitude.

In addition to providing an attitude reference for stabilisation of sensors, such a complementary DVS/AHRS combination provides automatic holding of helicopter position in the hover when integrated with an autopilot and radar altimeter. A digital air data system provides airspeed and altitude.

Terrestrially referenced navigation uses sensors such as ADF, VOR/DME or TACAN, LORAN C/D, OMEGA and future compatibility with GPS will be required. Sensor management with automatic selection and tuning of navaids is provided.

A navigation data-base of waypoints/targets, routes, special steering profiles etc. is entered using the Data Transfer Device and stored for flight planning and flight guidance purposes. In flight, navigation locations can be entered into the data-base for transfer after the flight to a de-brief facility.

The ICDS provides lateral and vertical guidance commands for enroute and mission/attack profiles via the autopilot.

Engine and airframe Health and Useage Monitoring sub-system

With the necessity of keeping the helicopters available for flight at all times and in all weathers, the health and useage condition of the engine, transmission and airframe must be closely controlled.

The ICDS monitors engine and transmission data and other airframe parameters for adverse trends and limit exceedances, and maintains a log of such occurrences, along with aircraft usage statistics and performance data. Parameters related to the life of engine and transmission components, such as torque, are monitored and exceedances down-loaded via the DTD.

Alerts, Warnings and Cautions with MFD and CDU message pages and scrolling checklists based on the flight manual are activated to assist the crew carry out normal and emergency procedures.

Performance management functions are provided with the ICDS monitoring fuel flow; mission profile data being entered using the DTD. The data available enables the crew to establish if the required mission endurance can be achieved,

Hardware features

Processor Interface Unit (PIU) / Symbol Generator

PIUs are rack mounted units forming the digital core of RAMS configurations. A PIU contains up to 18 modules for the processing and input and output interfacing functions. A variety of analog and digital modules interface the processing function to the rest of the avionics suite.

The range of interface modules fitted to the PIU, including MIL-STD-1553B, covers a wide range of communication and navigation equipment as well as providing various analog and discrete interfacing, software configured to match the other avionic systems.

Multi-function Display (MFD)

The MFD incorporates a high brightness, high resolution monochrome cathode ray tube (CRT) display and software controlled function keys around the lower 3 sides of the display area.

The video input is compatible with 525, 625 and 875 line standards, when displaying an external video input, alpha-numeric and symbolic data can be overlaid on the picture.

Control and Display Unit (CDU)

The CDU incorporates a monochrome CRT display, line keys and keyboard. Where possible, control functions are accomplished without the pilot having to take hands off the controls but the CDU can be used to conduct all normal flight planning / navigational functions, selection and frequency control of all radios and the control and operation of other sub-systems. The operator can select which function he wishes to address by using the dedicated keys on the CDU keyboard. Line keys at the sides of the display, with software controlled functions, enable the operator to select the menu or page for the information or control function required.

The CDU also displays video or symbology data from an external source or symbol generator.

The CDU and MFD are compatible with night vision goggles and all push button keys are designed to be operated by gloved hands.

Data Transfer Device (DTD)

The DTD is used to transfer data between a ground loading facility and the airborne equipment.

Communication frequencies, navigation waypoints and other tactical briefing data, as well as engine and transmission data, is stored in a battery maintained solid state memory.

Data is transferred in ASCII format from a Ground Loader Unit, via an RS-232C serial data link, to the DTD; which is then taken to the helicopter cockpit located Receptacle and the data transferred to update the data base in the PIU.

The data-base for the Ground Loader Unit may be updated by briefing personnel directly or via a data link.

Conclusion

For the integrated crew station of the battlefield helicopter, whether new build or retrofitted, an avionics management system configured as an Integrated Control and Display System can provide the following benefits,

Reduction in crew workload from the co-ordination of information and automation of system functions

Improvement in crew visibility by reducing cockpit panel area

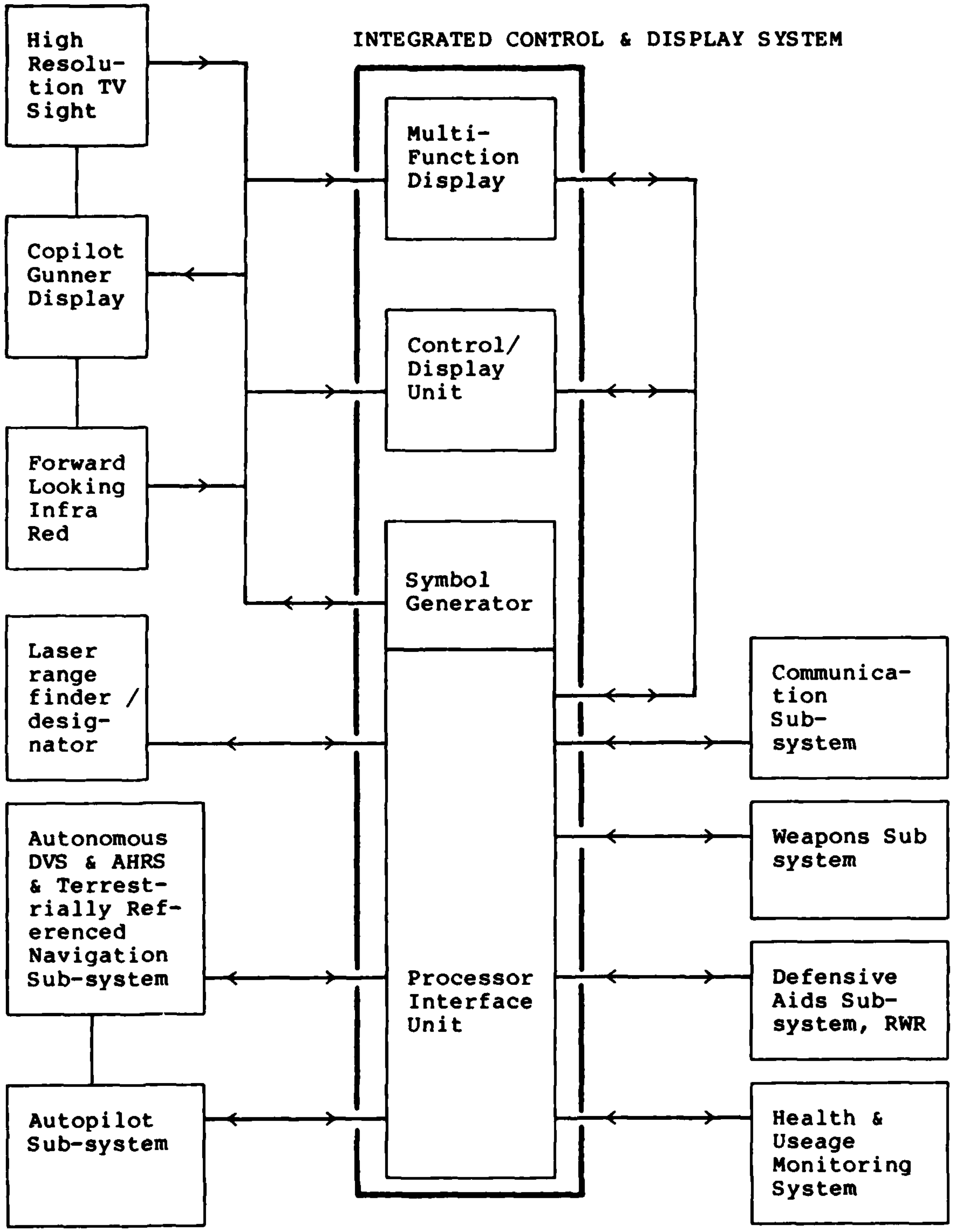
Reduction in overall system weight

Enhanced survivability with the crew having up to date knowledge of the tactical situation

Improved Reliability and Maintainability

Built-in capability for system expansion and enhancement

The experience gained from the systems integration and flight test of a Racal Avionics Management System configured for a multi-role light attack helicopter has proven the battlefield mission management concepts as described in this paper.



BLOCK SCHEMATIC OF AVIONICS SUITE FOR MULTI-ROLE LIGHT ATTACK HELICOPTER

SYSTEME NUMERIQUE DE PILOTAGE
POUR HELICOPTERE DE COMBAT

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RESUME

Depuis plusieurs années, la SFIM s'est spécialisée dans le développement des systèmes numériques de pilotage pour hélicoptère. Un environnement sophistiqué de simulation a été créé, qui permet de concevoir les systèmes futurs. Il a été utilisé au cours de ces dernières années pour la conception et la mise au point de systèmes qui ont fait l'objet d'une évaluation complète en vol sur plusieurs types d'hélicoptères.

Après avoir présenté les méthodes générales de conception et d'intégration de tels systèmes, nous parlerons successivement du système de pilotage numérique (PAN 1) couplé à une centrale d'attitude strap-down (26SH) et du système de couplage de vol stationnaire (HCL), lui-même connecté au système de visée (VENUS).

En conclusion, nous présenterons rapidement le système en cours de développement à la SFIM et qui est adapté à tout hélicoptère de combat.

1 - PRESENTATION DE L'APPROCHE SFIM EN MATIERE DE DEVELOPPEMENT DES SYSTEMES NUMERIQUES DE PILOTAGE

1.1. Généralités

Ce chapitre a pour but de présenter succinctement l'approche de la SFIM dans la conception et le développement des systèmes numériques de pilotage. Nous aborderons également la présentation des procédures de vérification et de validation associées ; ces procédures sont conformes aux recommandations de la DO 178 et reprennent certains compléments demandés par les Services Officiels Français.

Elles complètent l'approche méthodologique de la SFIM concernant la production des logiciels et des matériels. Elles correspondent d'une part à la spécification fonctionnelle initiale et d'autre part à l'intégration et à la validation finale d'un système tout au long de son développement, essais en vol compris. Cette approche est couramment utilisée à la SFIM dans le domaine des systèmes numériques de pilotage.

1.2. Spécifications formelles

Ce paragraphe regroupe les 3 notions suivantes :

- écriture des spécifications fonctionnelles établies à partir d'un modèle de simulation sophistiqué.
- écriture du document correspondant à la génération automatique des tests d'intégration du logiciel du système, à partir du modèle.
- écriture du document correspondant à la génération automatique des tests de validation du logiciel et du système lui-même, à partir du modèle.

1.2.1. Spécifications fonctionnelles

Elles sont établies à partir d'une simulation fonctionnelle complète du système.
Dans ce but, la SFIM a développé un environnement de simulation de système de contrôle automatique du vol, sur ordinateur PDP 11/70.
Ce type de simulation en boucle fermée est produite toutes les fois qu'un nouveau système est à l'étude. Le langage de haut niveau FORTRAN IV + est utilisé pour cette simulation.

Par ailleurs, la logique de fonctionnement du système en cours de développement est simulée dans le but de générer rapidement et facilement des profils d'évolution compatibles avec les contraintes opérationnelles.

Au niveau de cette phase de simulation, l'architecture globale et les connexions entre modules du logiciel de l'application sont figées et seront reprises telles que sur la configuration de développement des logiciels temps réel. Par ailleurs, toutes les constantes associées à chaque module font partie des spécifications formelles du système.
Il est important de noter que nous n'avons pas besoin d'un modèle très précis d'aéronef pour produire une telle spécification du système à partir de la simulation.

1.2.2. Document de test d'intégration du logiciel

Il est possible d'associer un jeu de tests dynamiques à chaque module spécifié comme nous venons de le présenter, au moyen de cette simulation fonctionnelle. Ces tests sont organisés en fichiers qui peuvent être générés très facilement en faisant fonctionner l'ensemble de la simulation non linéaire.
Les tests sont mémorisés, ainsi que les résultats prévus en simulation, sur des supports magnétiques divers (disques, bandes magnétiques, disquettes).
Enfin, ils sont regroupés sous forme d'un document qui décrit la méthode de génération des tests d'une part et les tests eux-mêmes d'autre part. Les tests sont représentés sous forme graphique ainsi que les résultats attendus, issus de la simulation.

1.2.3. Document de test de validation du logiciel et du système

Il a la même forme que le document de test d'intégration du logiciel. Les tests de validation du logiciel et du système sont générés de la même manière que les tests d'intégration du logiciel, à partir de la simulation fonctionnelle. Ils revêtent également un caractère opérationnel très important. Ils sont cependant envisagés à un niveau plus élevé relativement à la structure du logiciel de l'application.
Le nombre de tests à générer peut être très élevé. Les tests sont mémorisés, ainsi que les résultats prévus en simulation, sur des supports magnétiques divers ; ils sont transférés vers le banc de simulation temsp réel sur des bandes magnétiques de grande densité.

1.3. Phase de vérification

Cette phase suit le codage et le test unitaire du logiciel d'application, sur la configuration de développement de logiciel temps réel associée.
La phase de vérification, au sens où nous l'entendons, est préparée indépendamment, en utilisant les spécifications formelles décrites précédemment. Aucun test du matériel n'intervient dans cette phase.

1.3.1. Intégration du logiciel en temps différé

Dans les spécifications du système, nous avons vu qu'un ensemble de tests est associé à chacun des modules. Ces tests ont une signification opérationnelle dans la mesure où ils placent chacun des modules de logiciel propre au système dans une situation proche de la réalité.

La philosophie de cette procédure de vérification est la suivante :

a) A partir d'un fichier issu de la simulation en boucle fermée, il y a création d'un fichier de données d'entrée qui sont mises dans le format attendu par le logiciel d'application.

b) Ce fichier est transféré automatiquement vers la configuration de développement du logiciel temps réel.

c) Le logiciel temps réel est alors exécuté (en temps différé) en utilisant ces données et il y a création d'un fichier de résultats contenant les entrées/sorties et quelques variables internes à chaque module en cours de vérification.

d) Ce fichier de résultats est ensuite renvoyé automatiquement vers l'ordinateur de conception.

e) Finalement, les résultats attendus pour chaque test concernant des modules ou des fonctions à vérifier, et qui avaient été mémorisés sur l'ordinateur de conception au moment même de la génération des tests, sont automatiquement comparés à leurs homologues provenant de la configuration de développement du logiciel temps réel. Les résultats sont présentés sur console graphique, en superposition.

Toutes les différences significatives sont analysées et les erreurs corrigées. De tels tests sont très faciles à générer avec ce type de procédures automatisées et les résultats peuvent être analysés rapidement en relation avec les spécifications du système et les résultats de simulation.

1.3.2. Intégration des matériels et des logiciels assurant l'ensemble des interfaces (avec le pilote et avec l'aéronef)

Cette phase correspond à un fonctionnement en temps réel du système lors de l'intégration du logiciel et du matériel concernant les interfaces d'entrées/sorties (calculateur et poste de commande du système) entre les parties matérielles et les modules logiciels correspondant aux communications avec le monde extérieur.

Il est supposé, à ce niveau, que l'ensemble des ressources matérielles ont été correctement produites et testées, c'est-à-dire que le matériel est considéré sans pannes et plus particulièrement en ce qui concerne les aspects liés au temps réel et ceux liés à la sécurité.

1.4. Phase de validation

La phase de validation de l'aspect fonctionnel du système en temps réel est nécessaire pour compléter l'intégration du logiciel. Elle est également nécessaire pour éprouver l'opérationnalité du système complet inplanté dans le calculateur cible.

1.4.1. Validation de l'aspect fonctionnel du système

Pour cette phase de validation, des bancs de test temps réel sont utilisés. La configuration SILENE a été spécialement développée par la SFIM pour étudier les systèmes numériques de pilotage en temps réel.

Une analyse poussée de la fonctionnalité du système peut être effectuée en stimulant le calculateur cible à partir de profils d'évolution compatibles représentant l'aéronef en temps réel et son environnement associé.
Nous utilisons la même procédure que celle présentée pour la phase de vérification (§ 1.3.).

1.4.2. Caractère quasi-exhaustif des tests de validation du logiciel

Le nombre de tests envisagé peut être très important, de façon à stimuler le système automatiquement avec autant de profils opérationnels que possible.
C'est dans le but d'accroître la qualité des logiciels de pilotage par la génération de tests quasi-exhaustifs que la SFIM a développé tous ces outils et procédures automatiques. Ces procédures de vérification et de validation sont couramment utilisées, pendant l'intégration finale au laboratoire et pendant la période des essais en vol.

1.5. Traitement des modifications pendant les essais en vol

A chaque système de pilotage essayé en vol est associée une station de mesures SFIM. Pour les matériels embarqués, celle-ci est composée d'une unité d'acquisition microprogrammée qui acquiert tous les signaux à enregistrer (les entrées/sorties du système en cours de mise au point ainsi que certaines variables internes au logiciel d'application), selon un programme préétabli qui définit les cadences d'acquisition par rapport à un générateur de temps interne.
Ces signaux sont enregistrés sur un matériel du type enregistrement à cassettes auquel est associé un poste de commande.
Par ailleurs, la SFIM a développé dans son centre de conception un système de dépouillement automatique des cassettes enregistrées en vol.

Une telle approche permet :

. de définir les erreurs entre la théorie et l'expérimentation (remise en cause du modèle de simulation).
. de définir de manière rigoureuse les modifications à apporter au système (évolution des spécifications ou amélioration du logiciel).
. de rejouer en temps réel les essais en vol sur le 2ème prototype du système connecté en permanence à la configuration d'essais (SILENE).

2 - DESCRIPTION ET UTILISATION DE L'ENVIRONNEMENT DE SIMULATION

Ce chapitre a pour but d'insister sur le rôle joué par la modélisation d'un système de pilotage placé dans un environnement équivalent à l'opérationnel et ce, tout au long de la vie du projet.

2.1. Description de l'environnement de simulation

2.1.1. Généralités

Il s'agit de réaliser une simulation en boucle fermée de tous les éléments intervenant dans la conception d'un système complexe de pilotage automatique pour hélicoptère. La SFIM a développé les programmes en FORTRAN IV + sur PDP 11/70 concernant les modèles suivants :

- modèle d'hélicoptère avec la coopération de l'Aérospatiale-Marignane
- modèle des senseurs
- modèle des actionneurs
- modèle des perturbations
- modèle du système en développement

2.1.2. Modèle d'hélicoptère

Nous utilisons le modèle S80 (mis au point par l'Aérospatiale) qui est un modèle de connaissance fondé sur des bases physiques. C'est une description aussi fine que possible de la mécanique du vol et de l'aérodynamique de l'hélicoptère.

C'est un modèle universel qui permet de représenter n'importe quel hélicoptère à partir de ses caractéristiques physiques et aérodynamiques.

Pour chaque hélicoptère considéré, les caractéristiques en question sont recalées à partir des essais en vol, ce qui permet d'améliorer le comportement du modèle au cours du temps. Le modèle obtenu à l'heure actuelle est considéré comme correct dans tout le domaine de vol de l'hélicoptère, du stationnaire à la croisière.

Ce modèle est fortement non linéaire.

2.1.3. Modèle des senseurs

Au cours de ces dernières années, une modélisation de tous les senseurs intervenant dans un système de pilotage pour hélicoptère a été élaborée. A chaque senseur est associé un jeu de données caractérisant ses propriétés.

Ces caractéristiques sont issues du traitement de résultats d'essais en vol et de données techniques.

Grâce à une mise à jour des statistiques obtenues sur les systèmes déjà mis au point en vol ou en cours de mise au point, les coefficients associés à ces caractéristiques sont régulièrement recalés au niveau de la modélisation. Ce modèle des senseurs est connecté au modèle d'hélicoptère, qui définit les axes dans lesquels les mesures sont simulées.

2.1.4. Modèle des actionneurs

De la même manière, nous avons développé une modélisation des actionneurs classiques tels qu'on les trouve sur hélicoptère à l'heure actuelle :

- les servos-série à autorité limitée pour la sécurité
- les trims-parallèle, à vitesse maximale de déroulement limitée et à seuil de fonctionnement.

Ces organes sont fortement non linéaires et participent à la sécurité du système de pilotage automatique.

-les phénomènes parasites tels que les déformations des commandes de vol, les retards induits ... sont également modélisés à ce niveau.

Ce modèle des actionneurs est connecté au modèle d'hélicoptère sur les entrées de commande des 4 axes de pilotage.

2.1.5. Modèle des perturbations

L'environnement opérationnel dans lequel tout système de pilotage pour hélicoptère fonctionne nous a amené à simuler les effets de perturbations diverses qui influent considérablement sur les performances du système (hélicoptère + avionique).

Les principaux modèles développés à cet effet sont :

- une représentation de la houle en trois dimensions, pour simuler les vols au-dessus de la mer par mauvais temps.
- une représentation des rafales de vent et des phénomènes de turbulence.
- une représentation des principaux types de panne qui peuvent affecter le fonctionnement du système de pilotage.

Ce modèle des perturbations est connecté au modèle des senseurs qui mesurent leurs effets, vus du système de pilotage.

2.1.6. Modèle du système en développement

Une simulation aussi précise que possible du système de pilotage en cours de développement est alors effectuée. Il s'agit en fait de réaliser les documents de spécifications formelles (cf § 1.2.) à partir de la simulation fonctionnelle du système ainsi placé dans son environnement opérationnel, grâce aux modèles exposés précédemment.

2.2. Moyens de dialogue avec cet environnement de simulation

L'ensemble des programmes présentés précédemment tourne sur PDP 11/70 en FORTRAN IV +. Les fichiers correspondants sont mémorisés sur un disque de grande capacité (256 Mo). Les moyens d'entrée sont des consoles alphanumériques.

Les moyens de sortie se composent d'une console de visualisation graphique avec hard-copy, d'une table traçante, d'une bande magnétique et d'une imprimante rapide.

Par ailleurs, plusieurs programmes utilitaires ont été spécialement développés pour les besoins de la mise au point en simulation des systèmes de pilotage numérique. On distingue notamment :

- des programmes de tracé automatique sur la console graphique ou la table traçante.
- des programmes d'aide à la conception des systèmes pour la synthèse des correcteurs ou la synthèse de filtrages complexes.
- des programmes de formattage automatique des données relatives au système en cours de développement (réglages spécifiques).
- des programmes de formattage automatique des tests qui sont utilisés dans les phases de vérification et de validation du système (cf § 1).

2.3. Utilisation de cet environnement de simulation et possibilités

2.3.1. Dans la phase d'étude et de conception

a) Il s'agit dans un premier temps d'optimiser l'ensemble des algorithmes contenus dans le système de pilotage lui-même et de déterminer les différents réglages spécifiques grâce à la simulation en temps différé.

Les différentes lois de pilotage sont ainsi analysées. En particulier, l'analyse de la robustesse du système dans tout le domaine de vol de l'hélicoptère est facilitée par un tel environnement de simulation.

Il est ainsi possible de pré-déterminer toutes les constantes qui optimisent le système complet vis-à-vis des performances demandées.

b) Dans une seconde période, l'optimisation des calculs qui seront programmés ultérieurement dans le calculateur temps réel, est effectué.
L'objectif est double ; il s'agit d'une part de minimiser l'occupance du calculateur temps réel utilisé pour l'application et d'autre part de respecter les performances du système fonctionnant en boucle fermée sur l'hélicoptère.

2.3.2. Dans la phase de développement au laboratoire

Nous avons vu aux paragraphes 1.3. et 1.4. le rôle important joué par l'environnement précédemment décrit, notamment dans la phase de vérification du logiciel de l'application et dans la phase de validation du logiciel et du système lui-même. Ces procédures sont couramment utilisée à la SFIM et elles donnent d'excellents résultats.

2.3.3. Dans la phase des essais en vol du prototype

De plus, en complément de ce qui a été dit au paragraphe 1.5., l'environnement de simulation permet de rejouer les mesures faites en vol en différents points de la boucle complète. Cela nous permet de remettre partiellement en cause et d'améliorer les modèles décrits précédemment, bien que les recoupements entre les essais en simulation et les essais en vol soient toujours assez bons. L'utilisation importante de l'environnement de simulation permet par ailleurs de minimiser la durée des essais en vol au niveau de l'optimisation des lois de contrôle du vol pour un hélicoptère et un système de pilotage donné.

2.4. Expérience SFIM concernant les systèmes numériques de pilotage ayant suivi une telle approche

Plusieurs systèmes de pilotage SFIM, utilisant les techniques numériques, ont subis l'approche présentée dans les deux chapitres précédents. Les résultats ont été jugés très bons. Nous rappelons dans le tableau ci-dessous les principales applications et les temps de mise au point en vol pour les systèmes qui ont fini leurs essais.

SYSTEME	HELICOPTERE PORTEUR	ANNEE D'ESSAIS EN VOL	TEMPS DE MISE AU POINT EN VOL
CASM 2000	DAUPHIN 361-002	1981	25 heures
CASM 2100	DAUPHIN 365.F	1983	60 heures
PAN 1	ALOUETTE III	1983	37 heures
CAS 1000	PUMA 330	1983	5 heures
CDV 155	DAUPHIN 365-F	a débuté en Avril 1984	-

3 - APPLICATION DES METHODES PRECEDENTES SUR LES SYSTEMES PROTOTYPES (PAN 1 + 26 SH) ET (HCL)

3.1. Présentation du système PAN 1

Le système PAN 1 est un pilote automatique 3 axes entièrement numérique réalisant un grand nombre de fonctions de pilotage transparent et certains modes supérieurs.

3.1.1. Aspect fonctionnel

a) Le système PAN 1 peut être connecté à un ensemble de capteurs analogiques classiques, tels qu'on les trouve sur hélicoptère aujourd'hui, et également à une centrale d'attitude numérique strap-down 26 SH à sortie numérique A 429. Il pilote des servos-série sur les 3 axes (tangage, roulis, lacet) et des trims électriques en tangage et roulis.

b) Les modes suivants ont été optimisés d'abord en simulation, puis en vol, conformément à ce que nous avons présenté dans les chapitres 1 et 2 :

. stabilisations

- tenue d'assiette en tangage
- tenue d'assiette en roulis
- tenue du cap actuel par le lacet

. modes supérieurs

- tenue de vitesse air actuelle
- tenue d'altitude barométrique actuelle par le tangage
- acquisition et tenue de cap sélecté par le roulis/lacet.

. transparence

- la synchronisation des mémoires de référence, en tangage,roulis et lacet
- beep-trim, en tangage et roulis
- pilotage au manche contre les efforts,en tangage et roulis
- débrayage manche,en tangage et roulis
- beep plus manche, en tangage et roulis
- virage aux pédales, en lacet
- virage coordonné, par le lacet.

Ces modes ont été réalisés et testés avec capteurs classiques et avec centrale d'attitude numérique strap-down 26 SH. D'autres modes ont pu être envisagés grâce à la présence de cette centrale d'attitude strap-down, à savoir :

- l'augmentation de stationnaire (sans Döppler)
- la transparence particulière associée à ce mode
- l'augmentation de maniabilité en vol tactique.

Ils ont également été réalisés et testés.

c) Nous aborderons ici de façon plus précise les trois modes listés précédemment, sachant que ceux abordés auparavant sont assez classiques.

1) L'augmentation de stationnaire et la transparence associée

Ce mode se caractérise par la possibilité de tenir automatiquement le vol stationnaire à partir des informations fournies par la centrale inertielle 26 SH, et bien que cette dernière ne possède pas d'information de vitesse sol extérieure du fait de l'absence de radar Döppler.

Le but de l'optimisation d'un tel mode est multiple :

- essayer de diminuer la charge de travail du pilote par rapport à la tenue de stationnaire manuel ou la tenue de stationnaire avec le pilote automatique de base, et ce, pendant des durées moyennes voire suffisamment longues,

- avoir un mode de secours assez performant pour pouvoir tenir le stationnaire après panne du radar Döppler,

- avoir un stationnaire discret dans le cas d'un hélicoptère armé, par arrêt volontaire du radar Döppler en zone hostile.

L'objectif visé pour le temps d'intervention du pilote humain est de 5s toutes les 30 secondes environ, le but ultime étant des périodes de transparence très courtes et des périodes de recalage aussi longues que possible.

Il a été également envisagé d'initialiser plus correctement la centrale strap-down 26 SH en utilisant une mesure des vitesses-air à basse vitesse afin d'améliorer la qualité des vitesses-sol estimées (meilleures convergence des algorithmes d'estimation et meilleure précision des états estimés).

2) L'augmentation de maniabilité en vol tactique

De manière classique, pour éviter que le pilote automatique contre trop l'hélicoptère quand ce dernier est piloté en transparence par le pilote humain, la voie directe de la loi de pilotage est écrêtée. Cependant, en vol tactique, cette solution peut être améliorée. L'objectif de l'augmentation de maniabilité en vol tactique est l'obtention d'une vitesse angulaire fonction de l'effort exercé par le pilote humain sur le manche, c'est-à-dire fonction du déplacement du manche par rapport à sa position de repos et suivant la même loi dans tout le domaine de vol.

Ce mode a été optimisé et donne des performances correctes. Plutôt qu'un détecteur d'effort sur le manche, capteur jugé cher, il utilise un capteur de position de manche.

3.1.2. Aspect matériel

a) Le synoptique (fig. n° 1) présente l'organisation générale du système numérique de pilotage PAN 1 tel qu'il a été installé sur une Alouette III du C.E.V. de Bretigny.

On distingue notamment les différents capteurs analogiques qui ont été utilisés dans l'expérimentation en vol du système PAN 1, la centrale d'attitude strap-down 26 SH à sortie numérique, le système d'enregistrement des mesures en vol (UAM) connecté au système PAN 1 par liaison numérique.

On remarque par ailleurs l'organisation des commandes de vol (3 servos-série et 2 trims) et les boitiers de commutation associés normal/secours, ainsi que les différentes valises et outillages utilisés lors des essais au sol et en vol du système PAN 1 et connectés à ce dernier par liaisons numériques.

b) Le calculateur numérique PAN 1 est organisé autour d'un microprocesseur 16 bits SBP 9900 de TEXAS-INSTRUMENTS (Technologie I^2L). C'est un système monoprocesseur monitoré par des sécurités analogiques. Les commandes des vérins de trim et des servos-série sont réalisées de manière analogique.

Les provisions d'entrées/sorties relativement à l'unité centrale sont les suivantes :

- 16 entrées analogiques alternatives
- 16 entrées analogiques continues
- 2 entrées ARINC 429 (100 kbits/s)
- 1 entrée asynchrone RS 232
- 16 entrées logiques protégées sélectables (+ 28 V ou 0 V)
- 16 entrées logiques non protégées sélectables (+ 28 V ou 0 V)
- 8 entrées logiques TTL
- 1 sortie ARINC 429 (100 kbits/s)
- 1 sortie asynchrone RS 232
- 16 sorties logiques complémentaires (+ 28 V écrêté et filtré, 0 V protégé)
- 16 sorties logiques protégées (0 V)
- 8 sorties logiques TTL.

Elles ont été dimensionnées de manière suffisamment large pour les besoins de l'expérimentation.
La capacité mémoire est la suivante :

- 16 Kmots de mémoire EPROM
- 2 Kmots de mémoire RAM secourue.

NB : Les mots sont des mots de 16 bits.

c) Le poste de commande du système PAN 1

Il regroupe tous les moyens de contrôle (actions, visualisations) à la disposition du pilote. On distingue l'engagement et la visualisation des différents modes présentés au § 3.1.1.

- engagement des axes de pilotage séparément (T, R, L) ou simultanément (PA)
- engagement des modes supérieurs classiques (AS, ALT, HDG)
- engagement du mode augmentation de stationnaire (F1 - expérimental)
- provision pour l'engagement d'une autre fonction (F2 - expérimental)

On distingue par ailleurs la visualisation de certaines alarmes :

- HTT, HTR : hors-trim tangage et hors-trim roulis.
- FAULT : défaut de fonctionnement majeur, du pilote automatique numérique entrainant une déconnexion automatique et la reprise en main par le pilote.
- De même, on distingue la visualisation des positions des 3 vérins-série par l'intermédiaire de 3 galvanomètres pour la détection des pannes d'actionneurs.
- Enfin, la fonction "dimmage" permet d'ajuster l'éclairage des différents voyants associés aux fonctions. Les alarmes sont non "dimmable".

3.1.3. Aspect logiciel

Le logiciel du système PAN 1 respecte les critères de modularité et de programmation structurée. Il est décrit à partir d'un macro-langage structuré, orienté pilotage et utilisant le jeu d'instructions du microprocesseur SBP 9900 de TEXAS INSTRUMENTS. Cette bibliothèque de macro-instructions est utilisée pour l'ensemble des systèmes qui ont été réalisés dans cette technologie et qui totalisent environ 140 heures de vol effectif à l'heure actuelle. D'autres applications, en cours de développement, l'utilisent également.
Le logiciel du système PAN 1 a été vérifié et validé avant essais en vol, comme nous l'avons décrit dans le chapitre 1 du présent exposé. Il a été par ailleurs mis au point (ajustement des lois de pilotage) et validé en vol sur le porteur, conformément aux procédures abordées précédemment (§ 1.5. et 2.3.).

3.2. Présentation du système HCL

Le système HCL est un coupleur de vol analogique réalisant les fonctions de tenue de stationnaire Döppler - Radiosonde et d'asservissement sur une ligne de visée.

3.2.1. Aspect fonctionnel

a) Le coupleur HCL a été essayé et mis au point sur un Dauphin ayant un environnement tout analogique. Il est par ailleurs couplé au système de visée SFIM-VENUS.

b) Concernant le coupleur HCL, les modes de pilotage suivants ont été optimisés en vol :

- tenue du stationnaire Döppler automatique, par le tangage et le roulis.
- tenue de la hauteur de stationnaire Radiosonde par le collectif.
- asservissement sur ligne de visée, par le lacet.

3.2.2. Aspect matériel

a) Le synoptique (figure n° 2) présente l'organisation générale du système HCL tel qu'il a été essayé avec l'Aérospatiale sur hélicoptère Dauphin. On distingue notamment les différents capteurs utilisés dans cette expérimentation, ainsi que les connexions aux autres systèmes : (visée VENUS et pilote automatique 145 E).

b) Le calculateur analogique réalisant les fonctions énoncées au § 3.2.1. est constitué d'un boitier électronique pesant environ 4 kg.

c) Le poste de commande du système HCL

Il regroupe tous les moyens de contrôle (actions, visualisations) à la disposition du pilote.
On distingue l'engagement et la visualisation des fonctions suivantes :

- engagement général du coupleur par bouton poussoir
- 3 interrupteurs permettant d'engager les fonctions stationnaire Döppler (DOPP), stationnaire radio-sonde (ALT) et asservissement sur une ligne de visée (VIS)
- 3 voyants visualisant l'engagement des fonctions précédentes
- un potentiomètre de "dimmage", permettant de régler l'intensité lumineuse des voyants et du bouton poussoir

Ce poste de commande est complété par des boutons poussoirs situés sur les manches pilote et qui permettent de couper ou de réengager les fonctions par télécommande.

3.3. Résultats d'essais en vol

3.3.1. Concernant le système (PAN 1 + 26 SH)

a) Stabilisations de base

Dans un premier temps, nous avons cherché à comparer les performances des stabilisations de base sur les 3 axes, tangage, roulis et lacet, une fois les lois optimisées pour les trois cas suivants :

1) - Utilisation d'un gyroscope de verticale en soute type Newmark et d'un cap gyromagnétique type CG 130 (SFIM).
2) - Utilisation d'un gyrohorizon de planche de bord type H140 (SFENA) et d'un cap gyromagnétique type CG 130 (SFIM).
3) - Utilisation de la centrale d'attitude strap-down 26 SH (SFIM) à sortie numérique.

Le tableau ci-après résume les performances obtenues :

CAS ENVISAGE	CAS N° 1		CAS N° 2		CAS N° 3	
CRITERES RETENUS	Résiduelle de pilotage	Erreur statique maximale	Résiduelle de pilotage	Erreur statique maximale	Résiduelle de pilotage	Erreur statique maximale
AXE DE TANGAGE	± 0.4°	< 0.2°	± 0.2°	< 0.4°	± 0.2°	< 0.1°
AXE DE ROULIS	± 0.3°	< 0.5°	± 0.15°	< 0.5°	± 0.2°	< 0.5°
AXE DE LACET *	± 0.8°	< 2°	± 0.8°	< 2°	± 0.3°	< 1.5°

Note : * pas de trim de lacet.

Globalement, en performance, on constate que le cas n° 1 est inférieur au cas n° 2, lui-même inférieur au cas n° 3.

b) Transparence

La mise au point des modes de transparence n'a posé aucun problème.

c) Performances des modes supérieurs classiques

- Mode tenue de vitesse-air (A.S) : très bonne précision (± 2 kts)

- Mode tenue d'altitude barométrique (ALT) : performances correctes (± 25 ft) en régime stabilisé et en capture d'altitude à partir de vitesses verticales allant jusqu'à ± 500 ft/mn à l'engagement.

- Mode acquisition et tenue de cap sélecté (HDG): coordination parfaite à droite et accélération latérale résiduelle < 0.03 g à gauche, expliquée en partie par l'absence de trim de lacet. Son fonctionnement a toujours été cependant jugé correct. Afin d'améliorer encore les performances pour ce mode, un écrêtage sur l'assiette de roulis commandée fonction de la vitesse air a été introduit grâce aux possibilités offertes par la technologie numérique.

d) Performance du mode augmentation de stationnaire

En phase de stabilisation,nous avons obtenu des variations maximales de ± 0.1 m/s en tangage et ± 0.25 m/s en roulis.

3.3.2. Concernant le système HCL

a) Performances des modes supérieurs classiques

- Mode tenue d'altitude radio-sonde (ALT) : très bonne précision (± 3 ft en air calme ; ± 5 ft en air turbulent)

- Mode tenue de vitesses-sol nulles (DOPP) : assez bonne précision (en air calme < 1m/s en VX et VY)

b) Performance du mode asservissement sur une ligne de visée (VIS) : précision < 1° dans tous les cas de vol envisagés.

4 - EXTENSION VERS UN SYSTEME INTEGRE DE PILOTAGE POUR HELICOPTERE DE COMBAT

4.1. Définition du système de pilotage

Comme nous venons de le présenter dans le chapitre précédent, la SFIM a développé et mis au point plusieurs systèmes prototypes relatifs à la réalisation des modes de pilotage nécessaires à tout hélicoptère de combat, à savoir :

pour les stabilisations de base :

- tenue des assiettes à long terme en tangage et roulis, avec trims automatiques
- tenue du cap actuel, avec trim automatique
- toutes les fonctions de transparence (synchronisation des références d'attitude, beep-trim, pilotage au manche contre les efforts artificiels, débrayage manche, beep plus manche, virage aux pédales et virage coordonné, augmentation et contrôle de maniabilité).

et pour les modes supérieurs :

- tenue de vitesse-air actuelle
- tenue d'altitude barométrique actuelle
- acquisition et tenue de cap affiché
- tenue de vitesse verticale
- tenue de hauteur de stationnaire et d'altitude de croisière radio-sonde, à basse altitude par le collectif, avec trim automatique
- tenue de stationnaire automatique avec Döppler
- augmentation de stationnaire sans Döppler
- asservissement sur une ligne de visée
- approche MLS
- pré-commandes de tir canon.

Ces modes ont tout d'abord été optimisés sur un environnement de simulation très complet puis validés en vol à l'exception des pré-commandes de tir canon. Les performances associées aux essais en vol de ces systèmes ont été quantifiées. Nous avons à cet effet présenté les avantages de l'utilisation des techniques numériques. A l'heure actuelle, la SFIM est en train d'intégrer tous ces modes de pilotage dans une architecture de calculateurs numériques, avec les objectifs suivants :

- stabilisations de base 4 axes, duplex donc opérationnel après toute panne simple
- modes supérieurs sur les 4 axes, simplex mais avec reconfiguration en cas de panne sur les capteurs (poursuite de la mission)
- minimisation du nombre de calculateurs (optimisation poids/volume/coût, passant par l'optimisation des entrées/sorties nécessaires associées à chaque calculateur).
- performances maximisées grâce à l'utilisation des techniques numériques
- niveau de sécurité adapté à la mission de l'hélicoptère de combat : vol aux instruments, équipage minimal, vol à basse altitude et grande vitesse, fonctions d'aide au pilotage manuel (puissance maximale, VNE ...)

- poste de commande unique, regroupant l'ensemble des modes du système de pilotage mais double (un pour le pilote, un pour le tireur) ; la commande des fonctions de pilotage n'étant pas intégrée dans la notion de poste de commande multiplexé
- utilisation maximale des moyens de visualisation situés dans l'hélicoptère (écrans cathodiques) pour synthétiser l'état de fonctionnement du système de pilotage
- maintenance 2ème échelon intégrée, grâce à l'utilisation des techniques numériques et simplification de la maintenance
- uniformisation des outillages et des bancs de test relatifs à un tel système intégré de pilotage.

4.2. Application de l'approche SFIM à un ensemble de sous-systèmes intégrés dans un système d'armes plus vaste

4.2.1. Synoptique général du sous-système de pilotage organisé autour du bus 1553 et relations avec le sous-système senseur de situation et le sous-système de visée

Dans l'architecture du système proposé (figure n° 3) chaque calculateur PA (1) et PA (2) est connecté au bus redondant multiplexé 1553 où sont disponibles toutes les informations nécessaires au pilotage de l'hélicoptère dans la totalité des modes cités au § 4.1.

Les informations qui transitent par le bus 1553, proviennent :

- soit de concentrateurs de données, connectés au bus et qui sont chargés de la gestion du bus 1553 (normal, secours)
- soit d'une centrale d'attitude strap-down (par exemple la 28 SH-SFIM) qui transmet les attitudes, les vitesses-angulaires, les vitesses linéaires (sol et air) pour tout le domaine de vol, les accélérations linéaires et qui réalise les fonctions de navigation.
- soit du système de visée (par exemple le système VIVIANE de la SFIM) qui transmet ses ordres vers le système d'armes et le pilote automatique.
- soit d'autres systèmes (armement, radar Météo, ...)

En ce qui concerne les moyens d'action et de visualisation associés au système de pilotage, nous avons retenu la solution suivante qui nous parait la mieux adaptée à l'hélicoptère de combat.

Chaque membre de l'équipage, pilote et tireur, a à sa disposition un poste de commande spécifique du système de pilotage qui lui permet d'avoir accès à l'engagement de tous les modes sans restriction. Le pilote a la priorité sur le tireur. Ce dernier peut cependant récupérer cette priorité par une commande spéciale en cas de mise hors d'état du pilote.

Chaque poste de commande est connecté à un calculateur PA. En ce qui concerne les visualisations associées au système de pilotage, nous avons retenu :

- la présentation des alarmes visuelles spécifiques de reprise en main d'urgence sur la planche de bord, associées à des alarmes sonores.
- la présentation de l'état de fonctionnement du système de pilotage sur les écrans cathodiques EADI (pilote et tireur). Les informations disponibles concernent la répétition des modes engagés, le niveau de priorité (pilote ou tireur), l'état normal ou dégradé relatif au fonctionnement du système (cas de pannes), les compte-rendus des autotests et des tests prévol, les compte-rendus des tests de maintenance.
- la présentation des commandes élaborées par le système de pilotage sur les 4 axes, sur les écrans cathodiques EADI et EHSI, pilote et tireur, en mode couplé et éventuellement en mode directeur de vol.

Pour les fonctions autres que celles relatives au pilotage, (navigation, suivi du vol, optimisation du vol, météo, ...) des postes de commande multiplexés, un pour le pilote et un pour le tireur, semblent être une solution optimisée.

Enfin, les calculateurs PA (1) et PA (2) alimentent les actionneurs sur les 4 axes de pilotage : des servo-valves double-corps et des vérins de trim électriques double-moteurs.

4.2.2. Particularités du sous-système numérique de pilotage

a) Grâce à l'architecture interne, matérielle et logicielle, retenue pour chacun des 2 calculateurs présentés auparavant, le système ainsi défini permet d'atteindre un niveau de sécurité important en fonctionnement duplex, comme en fonctionnement simplex.

   En fonctionnement duplex, le système reste opérationnel après toute première panne et dans le pire des cas, l'hélicoptère conserve son attitude. Il peut même, dans certains cas de panne sur les informations fortement redondantes, permettre une poursuite de la mission avec des performances et un niveau de sécurité quasiment identiques.

   En fonctionnement simplex, il est passif sur toute panne et dans le pire des cas, l'évolution de l'hélicoptère reste limitée. De la même manière qu'en fonctionnement duplex, le système peut permettre la poursuite de la mission avec éventuellement des autorités de servos limitées sur pannes de certaines informations redondantes.

b) Concernant une technologie possible de réalisation de ce calculateur, signalons que la SFIM utilise aujourd'hui pour ses diverses applications les 3 principales familles de microprocesseurs pour lesquelles elle dispose des moyens de développement associés (TEXAS, INTEL, MOTOROLA).

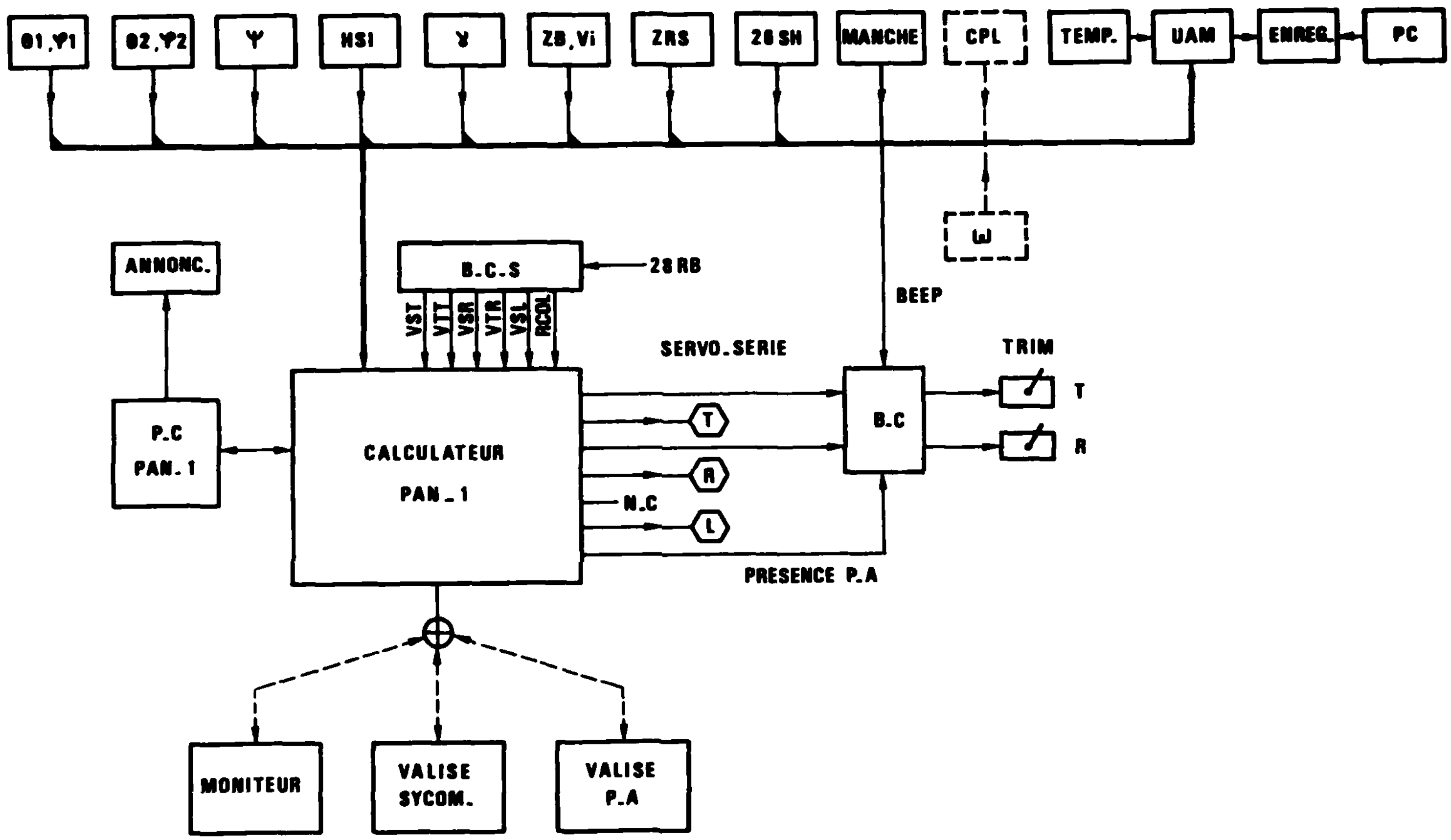

Figure 1 _ INSTALLATION DU SYSTEME NUMERIQUE (PAN1 + 26SH) SUR ALOUETTE III

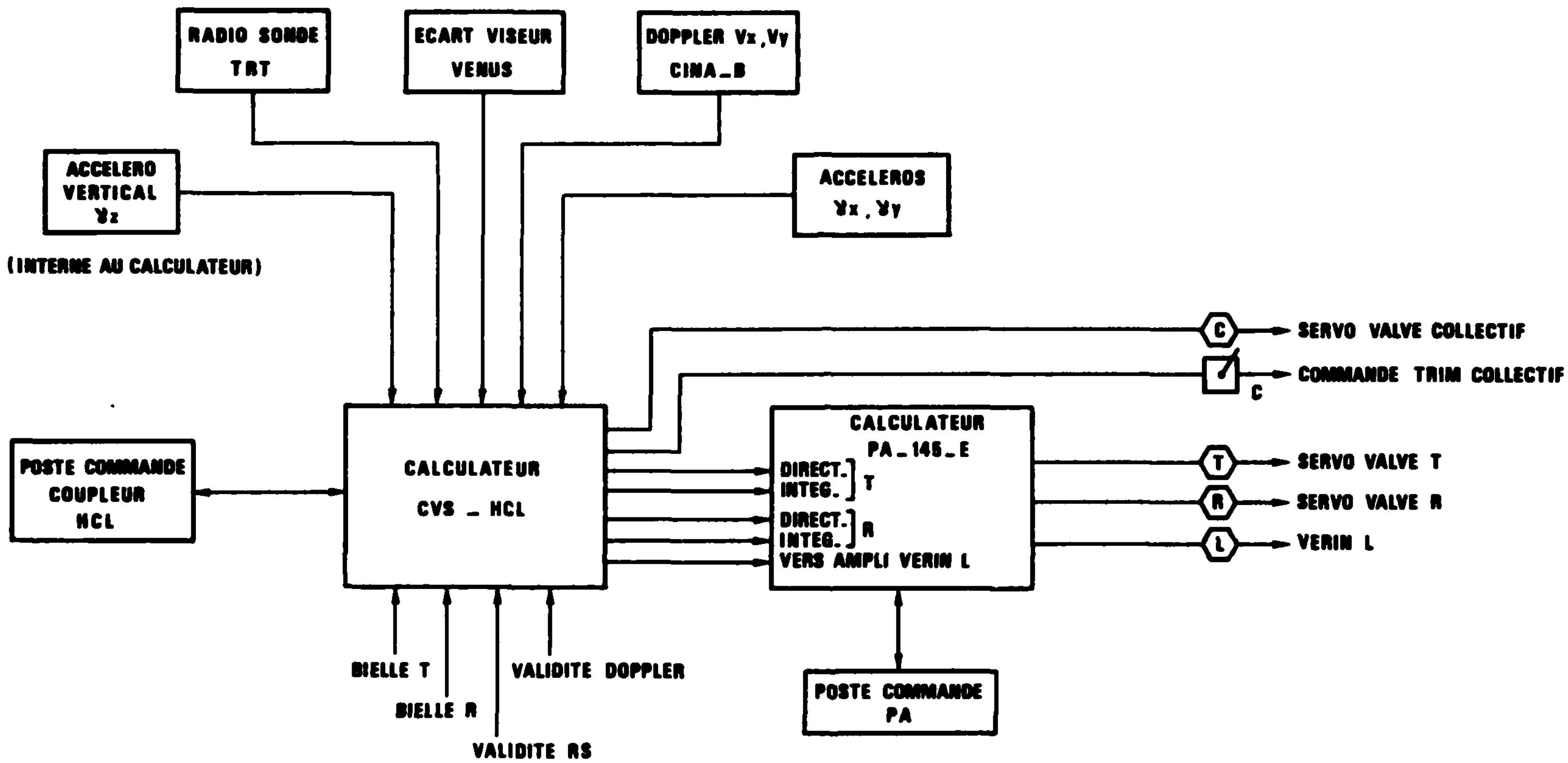

Figure 2 _ INSTALLATION DU SYSTEME HCL SUR DAUPHIN DE L'AEROSPATIALE _ MARIGNANE

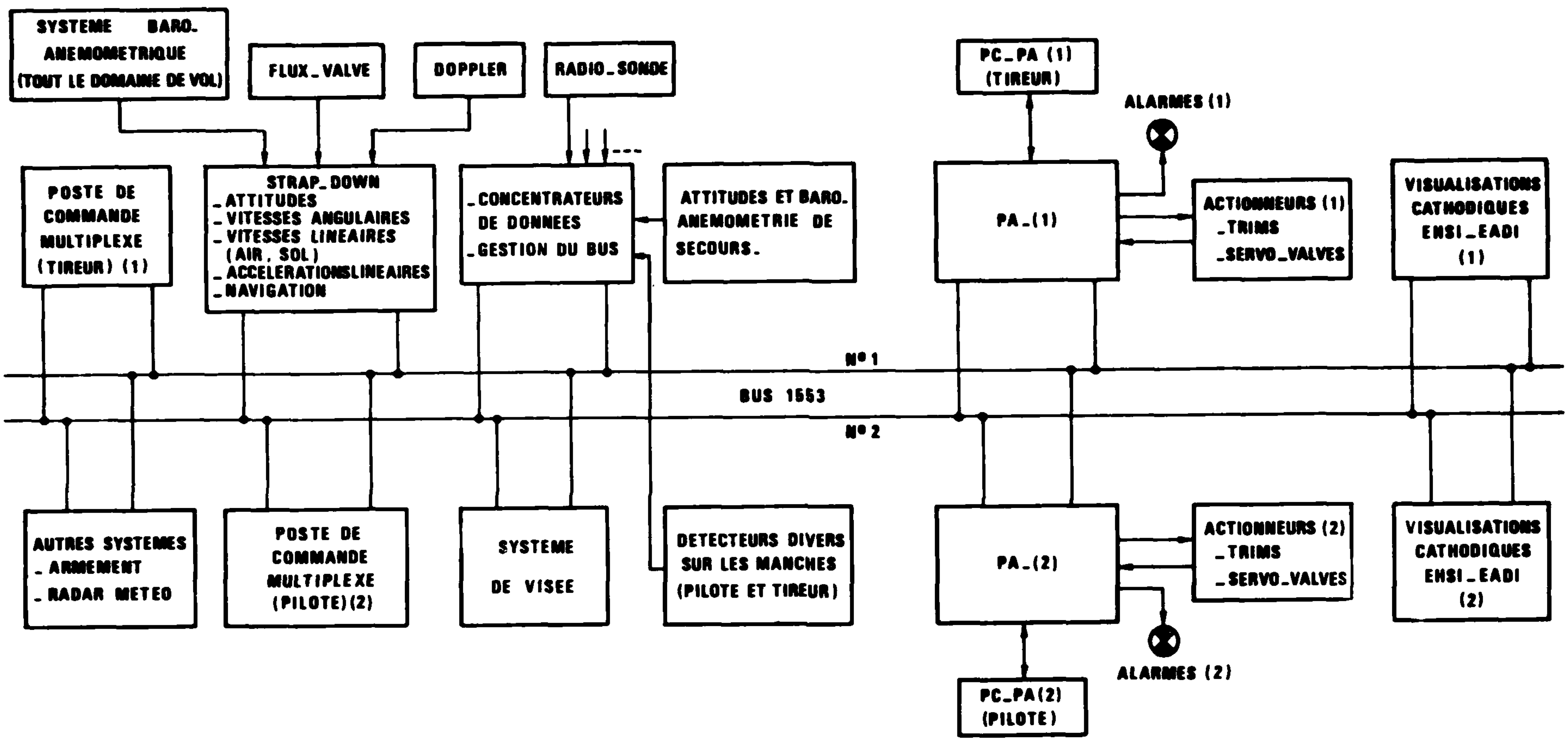

Figure 3 _ SYNOPTIQUE GENERAL DU SOUS-SYSTEME DE PILOTAGE ET RELATIONS AVEC LES AUTRES SOUS-SYSTEMES

TACTICAL HF COMMUNICATION FOR MILITARY HELICOPTERS USING THE NVIS MODE

by

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1. INTRODUCTION

The objective of this paper is to describe the experience gathered by the ELMER Company, on the use of loop antennas for tactical HF communications on board of military helicopters.

The helo platform is considered in either land or maritime scenarios.

2. BASIC CONSIDERATIONS ON THE USE OF THE HF BAND

To say that radio communication is vital for any military vehicle is only an understatement.

No mission can be considered effective without radio control or contact during the accomplishment and usually the mission results are strictly related to the efficiency of the communication system.

A crucial issue in the design of an efficient system is choice of frequency band, influenced by channel availability, propagation, power available, antenna gain, etc.

If we consider a semi-static base station and a moving vehicle, we find that due to antenna efficiency and spectrum crowding problems, the frequency region available to communication consists of the HF, VHF and UHF frequency bands.

VHF and UHF can offer line of sight communications over distances varying from a few miles in built-up areas to 200 miles for an aircraft flying at 60,000 ft.

However, there are tactical situations, in which line of sight conditions are not available, thereby precluding the use of V/UHF communications.

In this situation, two typical scenarios can be identified:

a) Land scenarios

The helicopter flies at low altitudes in rough or mountainous areas close to the "FEBA" line in reconnaisance or combat missions (Fig. 1).

b) Maritime Scenarios

The helicopter operates at sea at distances beyond the line of sight from the mother ship (Fig. 2).

In case a), communication with the cooperating units or the head quarters is vital, but at the same time low-level flying is necessary to reduce vulnerability to sophisticated antiaircraft weapons.

Case b) is typical of helicopters operating in ASW role with dunking sonar, which requires low altitude flying.

In both cases HF, either skywave or groundwave, can provide a solution to the communication problem.

HF groundwave communication improves in performance with decreasing frequency.

Its range at a particular frequency varies with surface conductivity, being greatest over salt water and minimum over dry sandy terrain; typical figures range up to 100 miles, but due to the strong dependance from the terrain features and frequency, ground wave propagation may prove to be quite uneffective or unpractical in the scenarios considered above.

Requirements may exist for large helicopters to communicate at sea over distances from the mother ship exceeding the range of reliable groundwave communication, taking into consideration the RF power constraints of helicopter radios.

Communication independent from the terrain features in the above scenarios is possible through an HF skywave propagation phenomenon whereby a radio frequency signal directed at a high angle of incidence (greater than 80°) within a limited band of frequencies, is reflected back to the earth from one of the several ionized layers producing an umbrella-type coverage.

The scientific foundations of this propagation mode, identified as NVIS (Near Vertical Incidence Skywave), have been known for quite a few decades, but use of NVIS did not receive widespread favor because of technical and operational problems, such as:

- difficulty in achieving a real-time prediction of the best operating frequency.
- lack of preset channelization in the available HF equipment.
- very low efficiency of the HF antenna.

The choice of frequencies to achieve NVIS skywave paths are different from those normally used for a long range circuit.

The usable frequencies lie in a small passbands (windows), 1 to 2 MHz wide, allocated in the 2 to 10 MHz frequency band.

The position of the frequency "windows" in the 2 to 10 MHz band depends on the time of the day, sunspot number and geographical location.

Recent experimentation has shown that in periods of high sunspot number, NVIS frequencies around 10 MHz can be used in the daytime while operation during periods of low SSN is possible at frequencies of 6 MHz and below; the same tests indicate that, once the path loss and environmental noise factors are overcome by the transmission system in a given geographical area, successful NVIS communication can be achieved at distance ranges up to 300Km irrespective of the terrain conditions.

Deviations from the optimum frequency results in an extra loss for the communication path (typically 10 dB for ± 1 MHz deviation).

To summarize the above considerations, HF communication over short range links can be achieved through ground wave or NVIS modes.

In some cases the HF groundwave channel may be adequate, but there are situations where NVIS is a better solution.

NVIS is definitely superior to groundwave over mountainuous or heavily forested terrains or over sandy soil. At sea, groundwave propagation is much more effective but the need for NVIS still exists for ranges exceeding 200 Km.

Today, real-time prediction of the frequency windows is much easier than in the past, due to the expanding use of computer facilities in the tactical environments.

The use of dedicated programs for the calculation of the available frequency band can be anticipated. In the design of the HF communication system for military helicopter it was decided that both groundwave and NVIS modes had to be retained.

Consequently a requirement was placed on the antenna system to provide satisfactory performance on both modes, i.e. a high angle coverage with predominantly horizontally polarized component for NVIS communication and the presence of a vertical component of the radiated electric field for groundwave propagation..

3. SELECTION OF THE ANTENNA

The HF antennas currently available for airborne installation belong to the three following families:

- open wire antennas
- loop antennas, including notches
- end-fed dipoles, including wing and tail caps.

This last category of antennas requires isolated parts of airframe in order to achieve efficient operation.

Wire antennas for HF communications traditionally consist of wires stretched between two points on the aircraft airframe or supported on short faired masts.

Some large helicopters have a multiplicity of masts supporting a long wire in a zig-zag arrangement.

The notch antennas consist of discontinuities in the aircraft metal structure, covered with dielectric radomes for aerodynamic purposes; the geometry and siting of the discontinuity must be such to achieve the maximum efficiency and the desired radiation pattern.

The loop antenna basically consists of a metal element supported by masts at some distance from the aircraft structure lying between the ends of the loop.

Both wire and loop antennas have two modes of radiation, namely:

- from the antenna alone;
- from the conductive portion of the airframe through energy coupled into it from the antenna.

It seems relevant at this point to review some of the most peculiar characteristics of the wire and notch antennas on one side and the loop antennas on the other side.

a. Wire and notch antennas

- Impose constraints on the aircraft structure; long wires are extended over a large part of the aircraft body.
- Broken antenna wires can seriously endanger the flight safety by interfering with control surfaces and rotors.
- Both wire and notch antennas feature very low radiation efficiency.
- The notch antenna requires an electrically insulated area in the aircraft structure.
- The radiation diagrams of wire and notch antennas feature strong variations with frequency.
- A coupler unit physically separated from the antenna is needed.

b. Loop antennas

- reduced size and weight
- mechanical robustness
- higher radiation efficiency when compared with other airborne antennas flexibility of installation
- coupler unit integrated in one of the antenna supports.

Fig. 3 provides plots of the efficiencies of a typical loop antenna and notch antennas of several aircraft types (courtesy of BAE, Bristol UK). Reference N° 2 contains a plot of the efficiencies of a vertical loop, horizontal loop and 12-foot wire antenna, shown in fig. 4..

4. OPERATION OF THE LOOP ANTENNA

Fig. 5.a is a simple model showing a loop antenna installed above a finite cylinder which simulates the aircraft structure.

Two modes of radiation can be identified, designated as the loop mode and the dipole mode

In the loop mode, current flow around the loop and the surface of the cylinder immediately adjacent produces a radiation pattern typical of a loop of larger physical size.

In the azimuth plane, vertical radiation arises from the vertical members of the loop and the pattern is essentially figure of eight with the minimum in the broadside direction (refer to fig. 5.b).

Horizontally polarized radiation from the horizontal member and its partial image in the cylinder has a maximum vertically upward.

Referring again to fig. 5.a, the section of the cylinder to the right of plane BB forms with the loop a Tee-match monopole which, combined with the section of the cylinder left to BB, forms an asymmetric dipole.

The radiation patterns will be those of a longitudinal horizontal dipole. Consequently, the combined radiation patterns in the azimuth plane will consist of a vertically polarized figure-of-eight with fore and aft maxima and a horizontally polarized figure of eight with port and starboard maxima.

The dipole pattern, shown as a dashed line in the azimuth plane, has a maximum vertically upward, due to the dipole asymmetry.

The relative amplitudes of these two patterns depend on the impedance which each presents to the feed point.

The loop mode resistance is a function of the fourth power of the frequency (f^4), while the dipole mode resistance depends on the sixth power (f^6).

Consequently, the loop mode predominates at the lower frequencies and the dipole mode at the higher frequencies. In the light of the considerations of paragraph 2, the choice of a loop antenna is thereby completely justified.

More specifically:

- the loop mode allows optimized ground-wave communication at the lower frequencies where the attenuation is lower (typically in the 2 to 4 MHz band) while still providing capabilities for the NVIS mode;
- the dipole mode provides efficient NVIS skywave communication at the higher frequencies (above 4 MHz), where "windows" are more likely to occur as demonstrated by experience.

The loop in fig. 4.a is shown as short circuited at the one end and series fed at the other.

Such an arrangement presents a low resistance and much higher reactance (due to the loop inductance) and necessitates a matching system, where the loop is tuned to resonance by capacitors.

The matching unit is a self-contained unit installed on the aircraft frame at one end of the loop antenna.

The loop resistance for a given periphery of the loop is inversely proportional to the diameter of the conductor and it is therefore advantageous to use as large a diameter as physical constraints permit.

The loop would ideally be mounted in the vertical plane through the aircraft fore-and-aft axis;

offsetting from the vertical reduces the vertically polarized component which is essential for groundwave propagation. The antenna may also be mounted on the side of the fuselage with its major dimension vertical.

This will give enhanced vertical polarization in the horizontal plane at the expense of the high angle radiation; a slant installation on the side of the fuselage is in most cases a good compromise.

5. STUDIES AND EXPERIMENTATION

The ELMER Company, in cooperation with the Italian Navy and AGUSTA COSTRUZIONI AERONAUTICHE, has conducted studies and experimentation on the use of HF loop antennas onboard small size (AGUSTA A-109 and A-129), medium size (AB-212 and AB-412) and large helicopters (SH-3D and EH-101).

The experimentation on the SH-3D helicopter has been specifically dedicated to comparison tests between the loop antenna, installed as indicated in fig. 6, and existing wire antennas (refer to fig. 7).

Measurements of groundwave radiated field at different azimuth angles have been conducted for the two antenna types, particularly at the lower end of the frequency range where the difference in the antenna geometry is more significant.

All the measurements have indicated a consistently better performance of the loop over the wire, typically a 16 dB to 8 dB improvement.

Skywave propagation tests at different distances and altitude of the helicopter have also been conducted; the results are summarized in the table below.

Distance (NM)	Altitude (feet)	Frequency (MHz)	Field intensity (dBm)	
			Loop	Wire
40	500	4.970	-86	-88
45	100	4.970	-86	-106
50	500	4.970	-80	-90
65	100	6.970	-91	-94
80	500	9.960	-96	-106
100	100	11.070	-96	-111

In this case too, the performance of the loop antenna is superior to the wire. A new area of investigation has been recently created by the MMI requirement for separate HF LINK-11 and voice communication facilities onboard the EH-101.

An experimentation has been recently completed by ELMER to investigate the electromagnetic compatibility of two loop antennas installed on the same ground plane.

The test results indicate that, by appropriate siting of the two antennas, simultaneous operation of the two HF transceivers is possible at a frequency offset of ± 10%.

6. CONCLUSIONS

The advantages provided in terms of reduced dimensions, increased efficiency and radiation characteristics, supported by the results of the ELMER experimentation, single out the loop antenna as an ideal radiator for use onboard aircraft and helicopters.

7. REFERENCES

1. HF loop antennas for air, land and sea mobiles by R.A. Burberry, British Aerospace Dynamics, Bristol UK.
2. HF antennas for airborne and surface vehicles by T.T. Walters, British Aerospace Dynamics Bristol UK.
3. ELMER Technical Report RT. 063 by G. Colafranceschi.

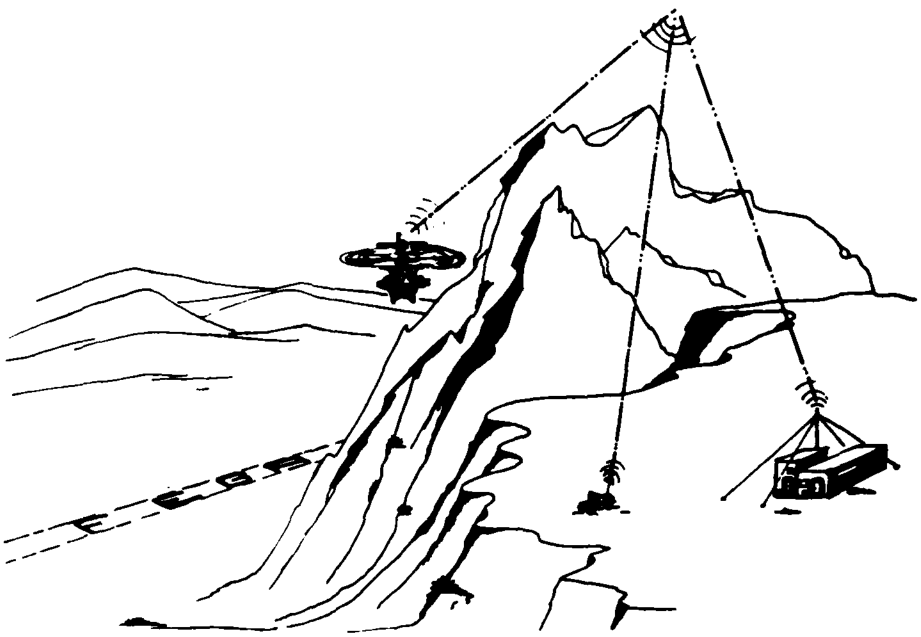


FIG. 1 - HELICOPTER FLYING AT LOW ALTITUDES IN MOUNTAINUOUS AREA

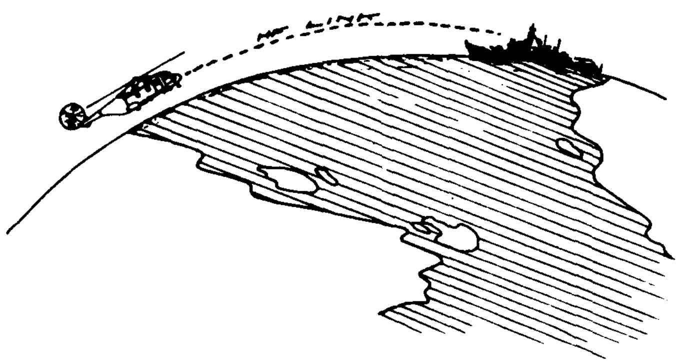
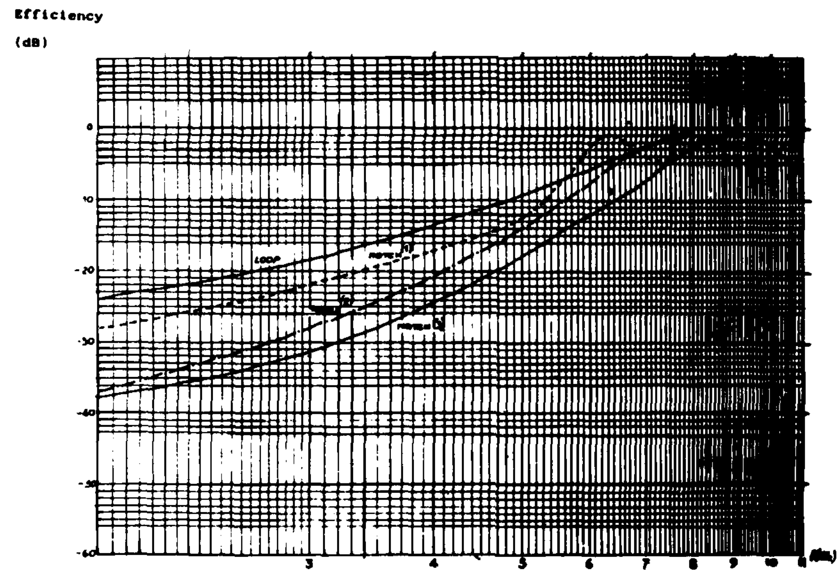


FIG. 2 - HELICOPTER FLYING AT LOW ALTITUDES OVER THE SEA



1) PHANTOM a/c   2) JAGUAR a/c   3) TORNADO a/c

FIG. 3 - EFFICIENCY VS FREQUENCY FOR LOOP AND NOTCH ANTENNAS

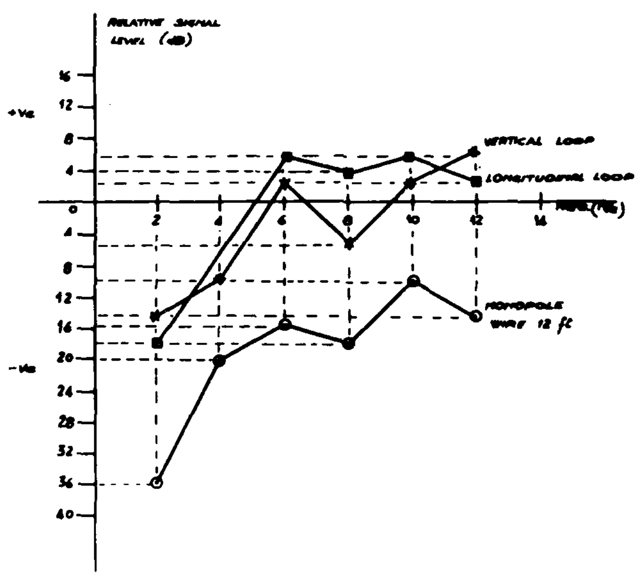


FIG. 4 - EFFICIENCY OF LOOP AND WIRE ANTENNAS

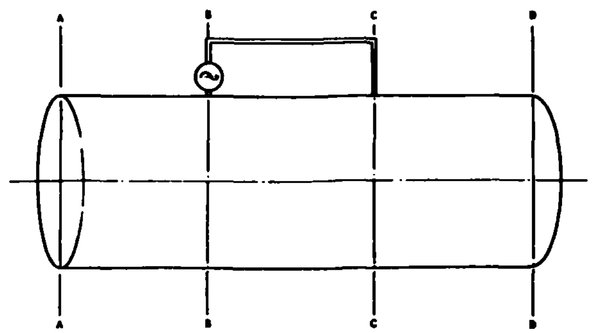


FIG. 5.a

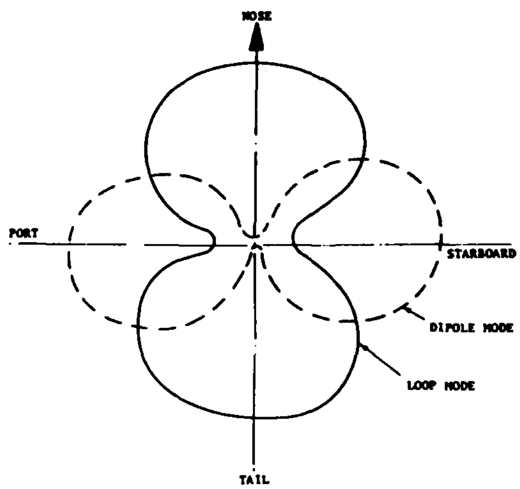


FIG. 5.b

FIG. 6 - EXISTING SH-3D WIRE ANTENNA

FIG. 7 - INSTALLATION OF A LOOP ANTENNA ON SH-3D

EVALUATION OF NOSE-, ROOF- AND MAST-MOUNTED SENSOR PLATFORMS FOR PILOTING AND SIGHTING, INTEGRATED IN FUTURE COMBAT HELICOPTERS

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SUMMARY

The piloting and sighting tasks under day, night and adverse weather conditions for a combat helicopter are different. For sighting the helicopter needs a well adapted sensor package, which may consist of a high performance FLIR, a direct view glass optic, a TV-channel and a tracker with a boresighting module. For the piloting task it is possible to use a complex sensor system including a helmet-mounted sight and display (HMS/D) and/or night vision goggles (NVG).

In this paper different aspects of nose-, roof- and mast-mounted sights (NMS, RMS, MMS) for a gunner will be discussed taking into account present and future weapon systems. This leads to advanced electro-optical systems with a high performance telescope, allowing an increased combat range. Some thoughts are given to an advanced multi-sensor approach where the more conventional systems are supplemented or partially replaced by a multi-mode imaging radar.

The sensor for piloting can be integrated with the sighting system or located in a position remote from the sight. Depending on the task, the pilot needs a dedicated display in the form of either a head-down display (HDD) or a helmet-mounted display (HMD). In either case, a superimposed symbology adapted to the phase of the mission e.g. cruise, transition or hover, is required.

Most of the solutions discussed are based on experimental results obtained during a number of flight tests with visionics systems on a Bo 105.

1. INTRODUCTION

The piloting and observation tasks under day, night and adverse weather conditions for a combat or a scout helicopter are essentially different.

The helicopter needs a well-adapted passive sensor package for sighting, which may consist of a high performance Forward Looking Infrared (FLIR), a Direct View Optics (DVO), a TV or Low Light Level TV channel (LLLTV) and a tracker with a boresighting module. A reconnaissance radar, laser range finder/designator (LRF/D) and a laser and radar warning system can be additionally integrated in the platform as active sensors.

It is possible to use a complex sensor system including a FLIR or a LLLTV with a wide field of view (WFOV) a Helmet Mounted Sight and Display (HMS/D) and/or Night Vision Goggles (NVG), for the piloting task (ref. 1, 2, 3, 4, 5).

The locations of various platforms for gunner/observer are discussed with their advantages and disadvantages in section 2. Vulnerability aspects for NMS, RMS and MMS, with measurements and calculations and the aerodynamic aspects are contained in sections 2.5 and 2.6 respectively. Section 3 gives details of a piloting sensor system, considering a gunner mast-mounted sensor as backup and rotor blade influences. Section 4 describes the aspects of a high performance electro-optical multi-sensor package for a modern gunner/observer system, with particular regard to weather conditions.

2. PLATFORM LOCATION

2.1 ASPECTS OF PLATFORM LOCATION

There are two possible seating configurations for a dual combat/scout helicopter, see Fig. 1

- o tandem configuration
- o side-by-side configuration

With the tandem version the gunner can sit either in the front or in the back, depending mainly on the location of the sensor platform. The tandem configuration has a smaller silhouette from the front than the side-by-side configuration. At present, modern combat helicopters use mainly the tandem configuration with a seat formation shifted by approx. 15° to 25° and a head radius of 10" from the eye to the window. The side-by-side concept will not be discussed further here. In a tandem configuration with a nose-mounted visionics system, including a DVO, the gunner sits in front. The seat positions of gunner and pilot will change, however, if a roof-mounted sight is used.

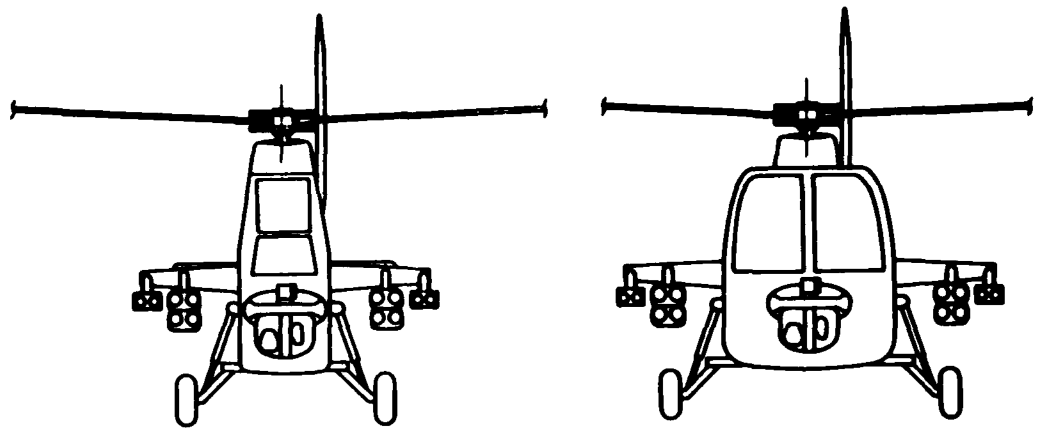

Fig. 1 Combat helicopter with tandem or side-by-side configuration

Other important aspects of the gunner sight location, which have to be taken into consideration, are:

- o the aerodynamic and flight mechanic influences of the platform shape
- o the field of view in AZ and FL including rotor blade influence
- o the accessibility of the sensor platform
- o the mass and centre of gravity, depending on high performance reconnaissance sight with good target acquisition, detection, recognition, identification, tracking
- o the available space for the platform
- o the platform vibrations
- o the visible, IR and radar signatures
- o the vulnerability including detectability, trackability, probability of hit, hit effectiveness (helicopter structure, redundancy)
- o the crashworthiness
- o the survivability (vulnerability and crashworthiness)
- o the weapon systems with self-defence capability

The sight must also conform to the operational and tactical requirements of the following weapon systems:

- o unguided chain gun (air-to-air, air-to-ground)
- o unguided, guided and fire and forget missiles (air-to-air, air-to-ground)

At present,three different platform installations are under discussion, which are dealt with within the following sections.

2.2 NOSE-MOUNTED SIGHT (NMS)

An NMS, including the platform, has to be designed to the helicopter fuselage, in such a way as to satisfy the aerodynamic requirements. In designing a helicopter the crashworthiness also has to be taken into consideration.

Fig. 2 shows a schematic representation of a combat helicopter with NMS for the gunner. The pilot sits in the rear. The nose-mounted TADS-PNVS (Martin Marietta Corporation) on the Hughes AH 64 is shown in Fig. 3.

The arguments for and against an NMS are presented below:

- o advantages of an NMS
  - o if the platform is mechanically well integrated into the helicopter structure, there are only limited aerodynamic influences. There is only a small reduction in max. speed
  - o a DVO can be used in the platform, if the gunner is seated fore
  - o the accessibility of sensors
  - o freedom of view in AZ and EL exists without rotor blade influence
  - o enough space is available for the platform
- o disadvantages of an NMS
  - o vulnerability (detectability, trackability, probability of hit and hit effectiveness) is relatively high since exposure time for weapon aiming/delivery is high
  - o the front silhouette during reconnaissance behind cover is large
  - o a heavy NMS can produce centre of gravity problems in a light combat helicopter
  - o the visibility for the pilot in the rear is reduced

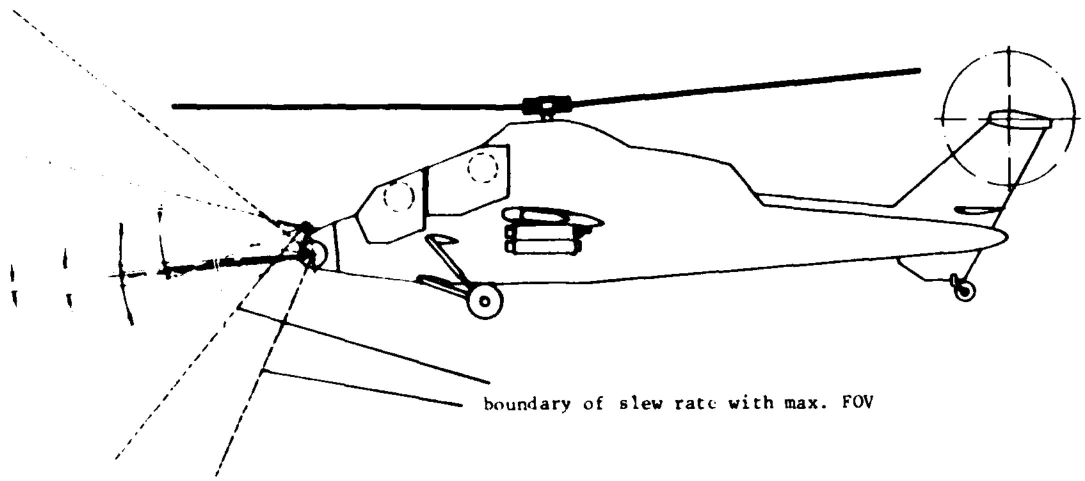

Fig. 2 Schematic view of a combat helicopter with a nose-mounted sight for the gunner (3 FOV). The pilot sits in the rear.

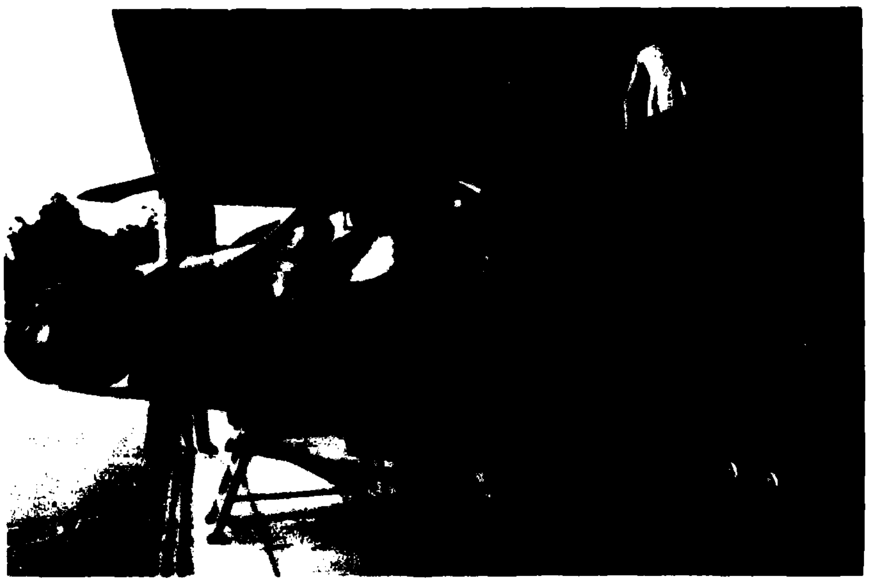

Fig. 3 The AH 64 with the nose-mounted TADS-PNVS, incl. DVO, TV, FLIR and LRF/D.

2.3 ROOF-MOUNTED SIGHT (RMS)

When a complex RMS is used, the helicopter roof structure has to be reinforced to carry the extra weight and to comply with the crashworthiness requirements. If the sight arm (down tube) is swivel-mounted, the gunner seated in the rear still has an unimpeded view for carrying out piloting tasks.

Fig. 4 shows a combat helicopter with an RMS for a gunner. He is seated in the rear and has a down tube and an eye-piece. The pilot sits in front and is provided with a nose-mounted FLIR as visual aid. A PAH 1 anti - tank helicopter with a roof-mounted DVO (APX 397 from SFIM) and side-by-side seat configuration is shown in Fig. 5.

- advantages of an RMS
  - a DVO can be used in the platform, the gunner sitting in the rear or in a side-by-side configuration
  - the accessibility of the sensor platform is adequate
  - the RMS produces minimal c.g. problems
  - the vulnerability with a reduced front silhouette from behind cover is reduced in comparison to the NMS, cf. section 2.5
  - the same is valid for reconnaissance from behind cover

o disadvantages of an RMS

- o the roof of the helicopter has to be reinforced
- o sufficient distance has to be allowed between rotor plane and sight
- o possible interference with air intakes
- o the view in EL is normally reduced in the downwards direction; the rotor blades may possibly exert an influence in the upwards direction, but the gunner only searches the sky in an air-to-air combat situation
- o the helicopter max. air speed is slightly reduced, especially if the equivalent drag area of the RMS is large
- o depending on the cover and the ammunition the helicopter is forced to "bob up" to fire

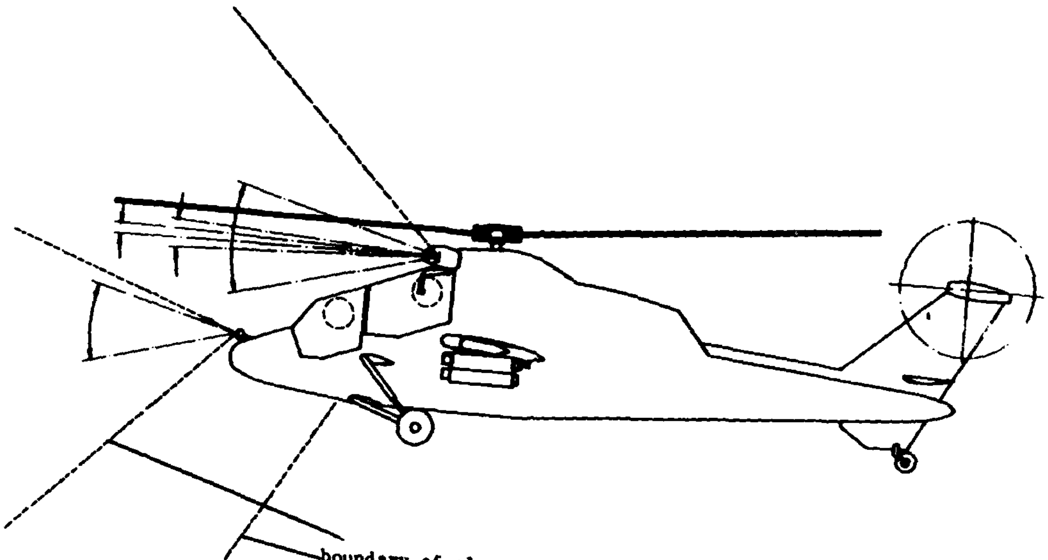

Fig. 4 Schematic representation of a combat helicopter with a roof-mounted sight for the gunner (3 FOV). The pilot sits in front.

Fig. 5 PAH 1 helicopter with a side-by-side seat configuration and a roof-mounted DVO day sight (APX 397 from SFIM)

2.4 MAST-MOUNTED SIGHT (MMS)

The mast-mounted sight has major advantages but also some disadvantages. The selection criteria of this system depend on the operational boundary conditions. The geometric dimensions of the MMS determine the periscopic view from behind cover.

Fig. 6 shows a combat helicopter with an MMS for a gunner. The gunner is normally seated in the rear and the pilot in front. The pilot has a nose-mounted FLIR sensor. In Fig. 7, the mast-mounted observation sight OPHELIA, installed on a Bo 105, is shown flying in front of trees.

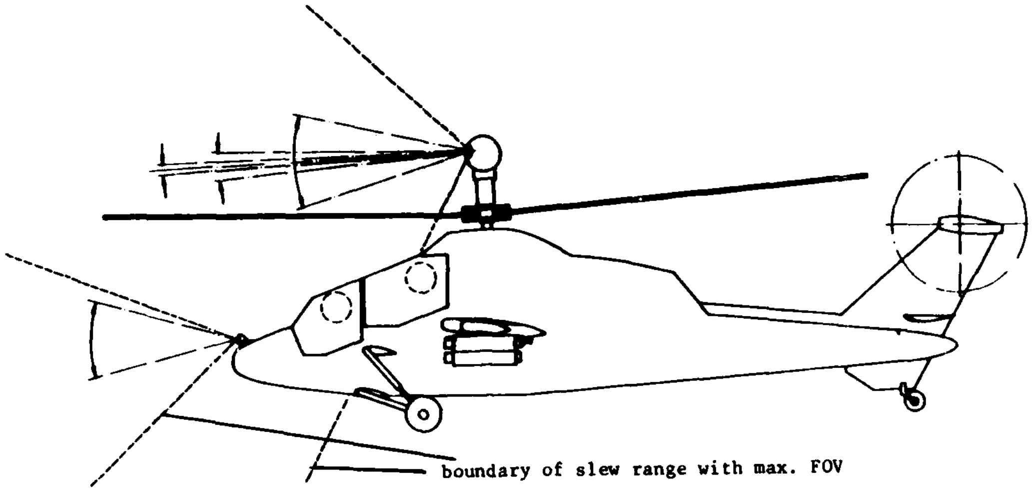

Fig. 6 Schematic representation of a combat helicopter with a mast-mounted sight for a gunner (3 FOV). The pilot has a nose-mounted FLIR sensor.

Fig. 7 The mast-mounted observation sight OPHELIA installed on a Bo 105 in a side-by-side configuration. An SMT FLIR (TRT) with two FOV, a TV channel and an LRF were installed in the 60 cm diameter stabilized steerable platform (SFIM).

- advantages of an MMS
  - the gunner has a periscopic view from behind protective cover, with an overall view of nearly ± 180° in AZ
  - target aquisition is possible without exposure from behind cover
  - the vulnerability is strongly reduced in comparison to both the NMS and the RMS, see section 2.5
  - the visible, IR and also radar cross-sections of the helicopter behind cover are relatively low
  - no influence on the centre of gravity in longitudinal direction
  - the exhaust gas of a missile does not disturb the optical window of the sight
- disadvantages of an MMS
  - through exposure of the sensor platform, the aerodynamic drag reduces the max. speed of the helicopter, see section 2.6
  - weight penalty arises to fulfil crash requirements
  - equipment accessibility is somewhat reduced
  - in general it is not possible to install a DVO, although a high resolution TV-camera can solve this problem
  - rotor blade interference occurs, if the LOS is directed downwards with a large angle of elevation, but this produces only a "chopper" effect

- o depending on the cover and the weapon, the helicopter has to bop up for weapon delivery
- o the cue identification of an MMS against the sky is easier in the visible and the IR spectral bands
- o problems may arise for the capture phase of a missile

2.5 COMPARISON OF NMS, RMS AND MMS WITH REFERENCE TO VULNERABILITY

In sections 2.1 to 2.4 the general advantages and disadvantages of NMS, RMS and MMS in a helicopter were discussed. In this section only the aspects of vulnerability with their criteria:

- detectability and trackability
- probability of hit and
- hit effectiveness (helicopter structure, redundancy)

will be taken into consideration for ground-to-air attack. The visible, IR and radar signatures are important f-ctors for helicopter detectability. In an operational mission, the combat/scout helicopter will fly n. of earth (NOE), using natural cover, until engagement. During NOE flight and hovering, cover in Central Europe is provided mainly by bushes, trees, small hills etc. The goal of an observation helicopter is to reconnoitre the battlefield without being detected itself. Target acquisition can be carried out with the unaided eye, but more extensively with the aid of electro-optical sensors which provide magnification.

Fig. 8 shows a helicopter hovering behind trees with a distance of approx. 5 - 10 m from the rotor tip to the trees. Three different front silhouettes of a helicopter with an MMS, an RMS and an NMS are seen. The helicopter should be fitted with hover hold (Automatic Flight Control System, AFCS), so that position is not affected by gusts. For NOE flight at night a good navigation system and a map reading device are essential. The ratio of the three front silhouettes refered to the cross-sectional area A is approx.

$$A_{MMS} : A_{RMS} : A_{NMS} \sim 1 : 2 : 5$$

These figures depend greatly on the helicopter dimensions and the position of the sights. If the helicopter changes its position to side-on, the surface ratio of the helicopter will be far worse for the NMS configuration.

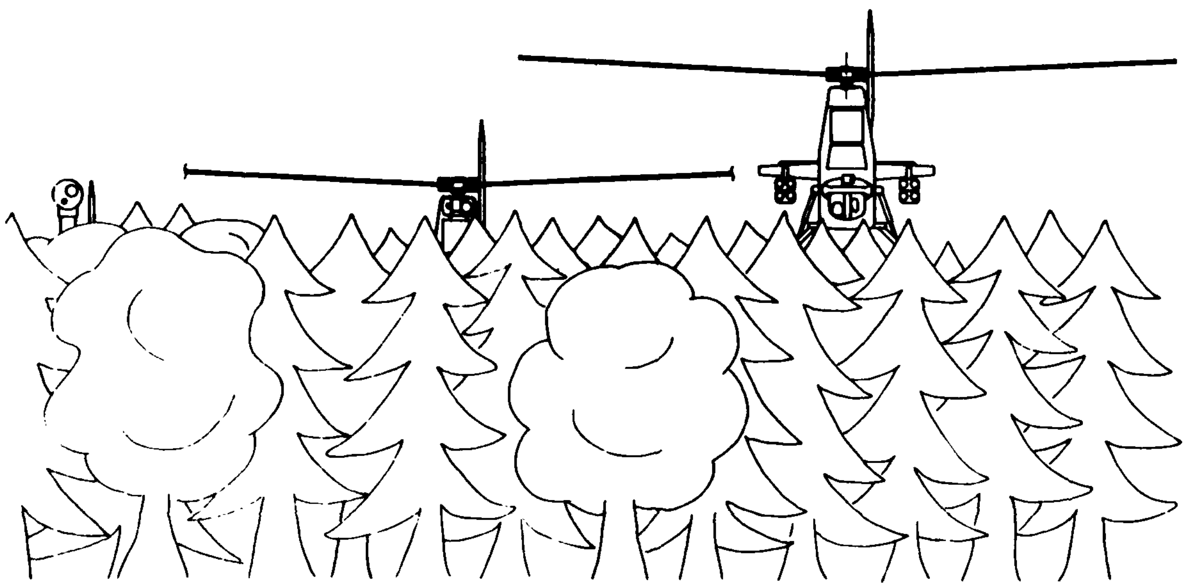

Fig. 8 Three different front silhouettes for an MMS, an RMS and an NMS helicopter hovering behind cover

In Dec. 1981 in Lichtenau (West-Germany) detectability comparisons were made by day between two helicopters of the same type - one with an RMS (PAH 1) and one with an MMS (Bo 105-OPHELIA) - related to visible and radar signatures observed from the ground. A helicopter with an NMS was not available. Two differently equipped tanks were used to try and detect the two helicopters. One tank had a DVO sight with 1x to 20x magnification, while the other tank had a search and tracking radar to register the differences in the radar cross-section between the RMS and the MMS helicopters. Unfortunately, owing to the poor weather conditions and the limited availability of the test helicopters, insufficient test data were obtained for a sound statistical evaluation.

The two stationary tanks were allowed to search for and observe the two helicopters separately for max. 3 minutes. 10 different types of cover were used with distances from the tanks ranging between 2 and 4 km, see Fig. 9. The observation time for the tank with the DVO was initiated when one of the two helicopters broke cover. The radar equipped tank was given an additional 30 seconds for target acquisition prior to the start of the 3 minutes. During this additional time, the commander could observe his display with the existing information. Thus, when the helicopter took up its hover position, the tank radar could detect the helicopter almost immediately because of the additional information on the display. The experimental conditions were therefore somewhat misleading.

Only few results were obtained, because the visibility in December was poor owing to heavy snowfall. The experiments showed a tendency to much lower detectability of a MMS helicopter compared to the RMS helicopter when both are observed with a DVO. An NMS-equipped helicopter should further substanstiate this tendency.

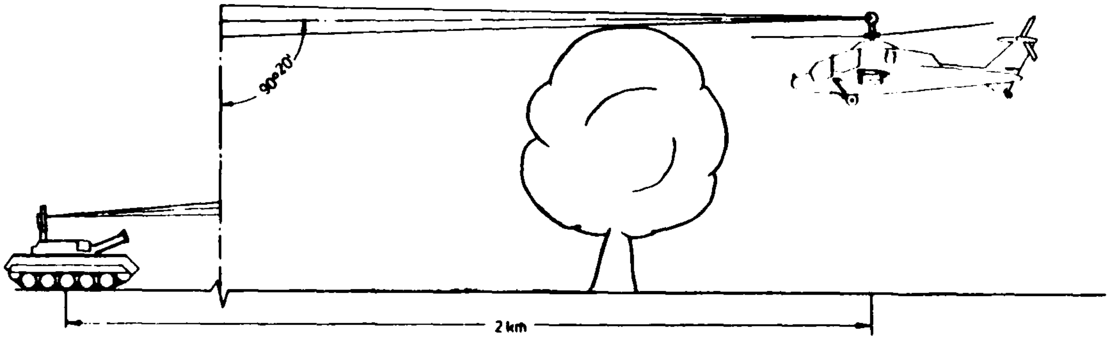

Fig. 9 A combat helicopter with MMS in the hover behind a tree observed by radar of a tank at 2 km distance

In the case of the tank with the radar system, the difference in detectability was not so well defined. This can be explained partly by the experimental conditions and partly by the stormy weather, the MMS helicopter having no hovering stabilization. Fig. 10 shows the results of the detectability and trackability measurements with the DVO and the radar system respectively as functions of time.

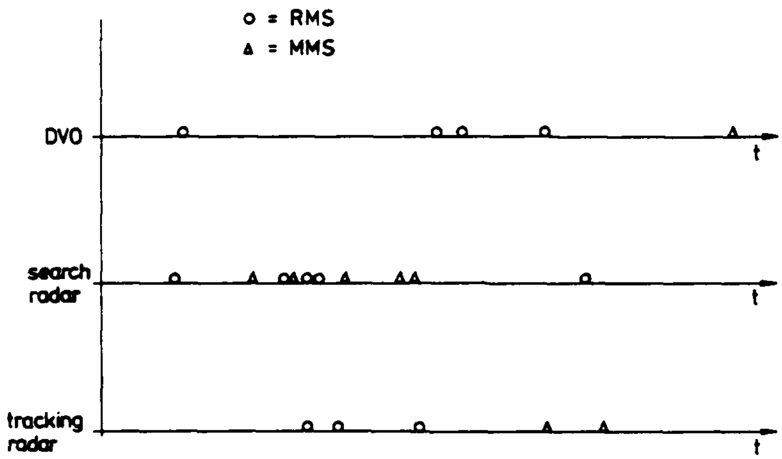

Fig. 10 Detectability and trackability measurements of RMS and MMS helicopters with a DVO and a radar system (ground to air) as functions of time. Five measurements for each helicopter (ref. 6).

The MMS helicopter was detected using the DVO in only 1 out of 5 experiments while the RMS helicopter was found on 4 occasions. Both helicopters were detected by the search radar in the max. available search time of 3 minutes in all 5 cases. The tracking radar needed more time for the tracking process, especially for the MMS configuration and was only successful in 3 or 2 out of 5 cases respectively. In addition the lock-on onto the helicopter with MMS was somewhat questionable. If the helicopter is itself fitted with a radar warning system, the tracking time would have been too long for a hit. The hovering helicopter would have been able to react to the threat.

Vulnerability calculations for NMS, RMS and MMS helicopters from a ZSU 23-4 tank, with 23 mm projectiles and equipped with a search and a tracking radar, have shown that the probability of hit for a MMS helicopter is reduced when compared to RMS and NMS helicopters. In these calculations detection and successful tracking of the MMS, RMS and NMS helicopters is presupposed. In Fig. 11 the probability of hit versus the slant range is shown graphically. In this figure the offset range of the tank was 735 m. Two other calculations were made with offset ranges of 1300 m and 10 m. A 10 m offset denotes flight over the tank. These different offset ranges simulate the changing helicopter cross-section and the angle subtended by the helicopter, behind cover, to the tank. Another assumption in all the calculations is that the helicopter has an AFCS and is hovering with nearly constant height above cover to observe the tanks. Only the cross-sectional area changes with the offset angle.

Realistic assumptions were made for the calculations concerning muzzle speed, damping constant, ballistic distribution, error of alignment in AZ and EL, cadence and time of projectile flight. The probability of hit for the MMS helicopter with AFCS is zero in the calculations. This probability increases considerably (e.g. approx. 20 % for vertical motion of $\pm$ 1.2 m in τ = 12 s) for all three versions in strong wind conditions without an automatic height hold.

The detectability, ability to be tracked and probability of hit for an MMS helicopter with an AFCS is reduced in comparison to RMS and NMS helicopters by more than 80 %. When the vulnerability of an MMS helicopter is improved of course the survivability, which includes the crashworthiness, will also be improved. A natural camouflage for the MMS fuselage is given behind cover. Results of comparisons made in the USA with two systems (MMS and NMS) show similar results, see ref. 7.

The hit effectiveness is not easy to calculate. Many parameters are included in this term, particularly the structural strength of the helicopter.

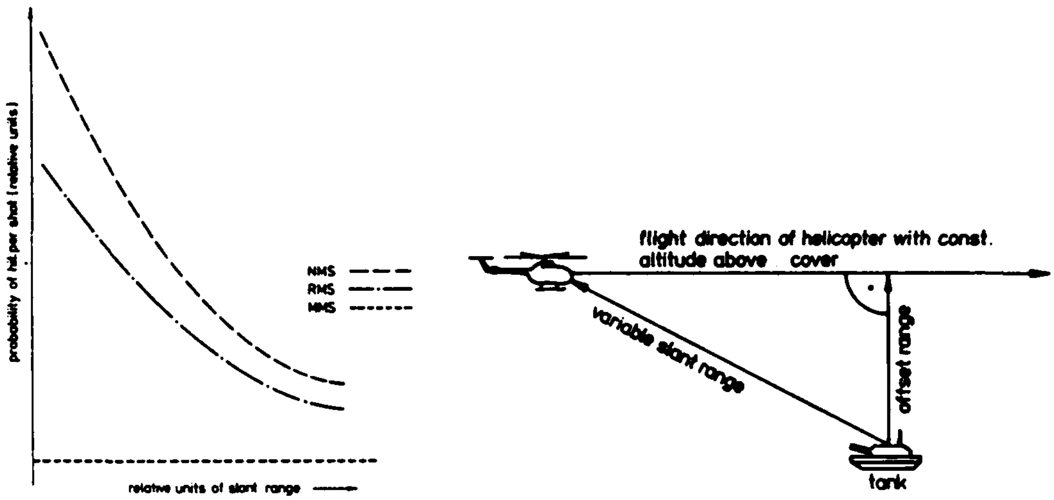

Fig. 11 Probability of hit per shot with successful tracking of an MMS, an RMS and an NMS helicopter (with AFCS) at 735 m offset range from a ZSU 23-4 tank with 23 mm projectiles versus the slant range

2.6 AERODYNAMIC DRAG OF A SENSOR PLATFORM

Theoretical investigations were performed in support of the experimental MMS-OPHELIA program installed on a Bo 105 (ref. 8, 9 and Fig. 7). Owing to the mast extension and sphere housing the optical equipment, the aerodynamic drag of the total helicopter is increased. Fig. 12 shows the Bo 105 power requirements and engine capacity for the equivalent test height corresponding to a density altitude of z_σ = 5000 ft (ISA). For comparison purposes, flight test measurements for the standard Bo 105 are given in the diagram and demonstrate good agreement with the theoretical predictions. Aerodynamic measurement with the addition of MMS (OPHELIA) indicate an increase in equivalent drag area estimated to be approx. 0.6 m². This is due in part to the greater pitch attitude trim angle and to the cabin roof modification, required to house the HUD, adding approx. 0.2 m², together with an estimated 0.4 m² for the mast extension (tubes) and sphere mounted on the rotor head. The additional drag was found to reduce the max. continuous forward speed of the Bo 105 with an MMS to 107 KIAS.

Calculations have shown that for a helicopter with max. speed of 280 km/h an increased drag area of 0.1 m² reduces the max. speed by 4 km/h (37 kW). The ratio of the three sights to the drag area F is approx.

$$F_{MMS} : F_{RMS} : F_{NMS} \sim 5 : 2 : 1$$

Other flight mechanics aspects e.g. trim, controllability, stability, rotor mast moments and vibrations have to be taken into consideration in NMS, RMS and MMS designs. On the Bo 105 no negative effects besides the reduction of the maximum horizontal speed were observed for the MMS.

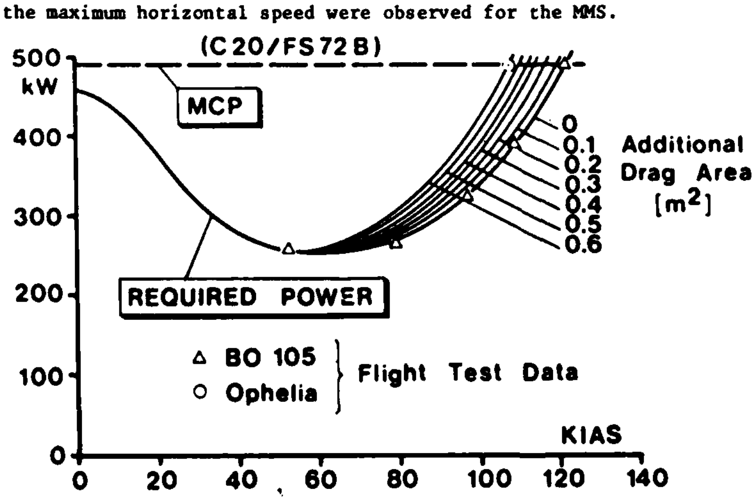

Fig. 12 Horizontal flight performance of Bo 105 and OPHELIA (MMS)

3. NOSE-MOUNTED PLATFORM WITH OPTICAL SENSORS FOR PILOTING TASKS

3.1 GENERAL REQUIREMENTS

Flight trials with a Bo 105 helicopter equipped with a pilot visionics system (PVS) have shown that a steerable platform is required for the optical sensors (FLAB program, ref. 1). Stabilization of the platform with a WFOV sensor (~ 30° x 40°) is not necessary, but stabilization can lead to better image results. The platform should be steerable over a range of AZ = ± 120° and EL = + 20°, - 50°. The best location for a piloting system is the helicopter nose. If the pilot looks down at an angle of - 50°, his FOV is not obstructed by the helicopter structure.

A nose-mounted platform with two optical sensors is shown in Fig. 13. The platform contains a FLIR (25.5° x 38°) and an LLLTV camera (30° x 40°) for comparison. To reduce the work load of the pilot, the platform was successfully steered with a helmet-mounted sight (HMS) using electro-magnetic principles. A beeper switch mounted on the collective stick was not found to be a good solution for platform steering.

Fig. 13 Bo 105 with a nose-mounted platform called pilot vision system (PVS) which includes two optical sensors, a piloting FLIR (IR 18 Mk II) and an LLLTV camera

The presentation of the sensor images is possible with either a head-down display (HDD), a head-up display (HUD) or a helmet-mounted display (HMD). Flight trials have shown that, for piloting tasks only, an HDD or an HMD give satisfactory results. Superimposed flight symbology with separate brightness control of the sensor image is desirable. The flight symbology has to be adapted to the phase of the mission e.g. cruise, transition, hover and bob up.

The pilot in a combat/scout helicopter needs a wide FOV FLIR with high thermal resolution (and/or LLLTV camera) for missions at night and in adverse weather conditions. At night and especially at dawn (around 04.00) the thermal contrast is generally very low. Temperature inversions with crossover points ($\Delta T = 0$ K) may exist which can be dangerous for flight decisions. The presentation of contour information by a FLIR is not as easy to interpret as in a visual band sensor. Sometimes the interpretation of a thermal image can be misleading. The estimation of distance in a video image is not very easy. The thermal resolution of $\Delta T \cong 0.05$ K with a FLIR is necessary to get a better image contour with low scene contrast. Blooming effects, which exist in an LLLTV and, in reduced form, in an NVG image, are not significant in a thermal image.

The best solution for a pilot visionics system, particularly for night and adverse weather conditions, is a combination of a thermal, an LLLTV and an NVG image using image superimposition. At the moment, such a system is not available as the NVG provide no direct video output. A combination of a FLIR (8-14 µm band) and an image intensifier e.g. NVG (0.6 - 0.9 µm) without superimposed images seems a good solution for military piloting tasks at present.

Figures 14 and 15 demonstrate two thermal images of two different FLIR's under good weather conditions with flight symbology superimposed.

Fig. 14 Thermal image (IR 18 Mk II, 25.5° x 38° with 4 Sprite detectors) with superimposed Cruise symbology during a night flight. Motorway intersection Munich-Salzburg

Fig. 15 Thermal image (PISA, 30° x 60° with 8 Sprite detectors) with superimposed Cruise symbology during day flight. A valley and trees producing definite shadows can be seen. 597 ft AGL, 53 kts airspeed, torque values T1, T2 = 40 %, 36 % and heading 330°.

3.2 REDUNDANCY CONSIDERATIONS

If the pilot FLIR or another sensor has a failure, the pilot can use the WFOV channel of the gunner FLIR as back-up. This is also true for the case of an MMS configuration. Owing to parallax, the LOS has then to be moved automatically in the corresponding direction of the pilot's FLIR before the failure, in order to view the surroundings at a short distance from the blade tip, see Fig. 16. The size of the MMS FLIR image increases in proportion to the parallax.

Rotor blade interference in the image occurs with a downwards LOS depending on the geometry of the helicopter. An example of a stationary thermal image is shown in Fig. 17. The rotor blade influence depends on the following parameters (ref. 9):

- o scan speed of the FLIR mirrors or polygons
- o scan mechanism (parallel, serial or serial-parallel)
- o field rate (CCIR, EIA)
- o FOV
- o rotor speed
- o distance of the MMS FLIR from rotor disc
- o rotor blade chord
- o number of rotor blades and
- o LOS angle of the FLIR through the rotor blades.

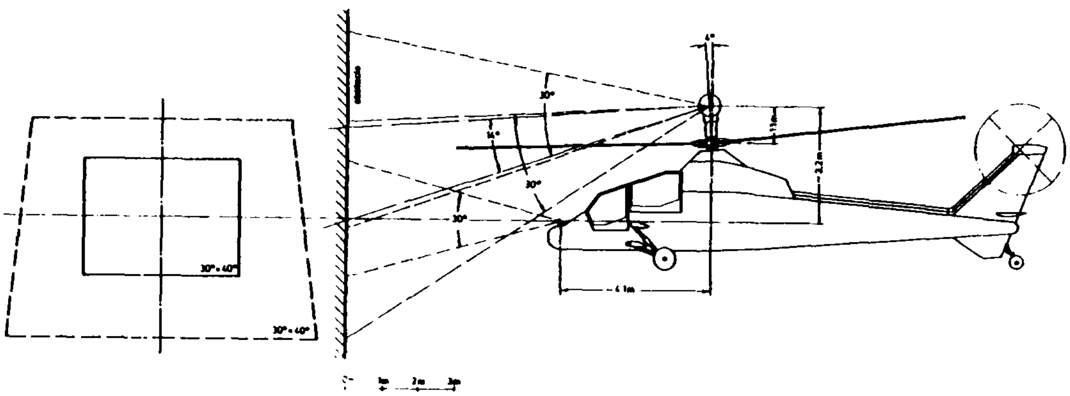

Fig. 16 Comparison of the pilot FLIR LOS (30° x 40°) in the nose version with the large FOV of a mast-mounted FLIR version, if the helicopter hovers in the proximity of an obstacle or cover. The angular correction depends on the geometry of the helicopter and the FLIR installations.

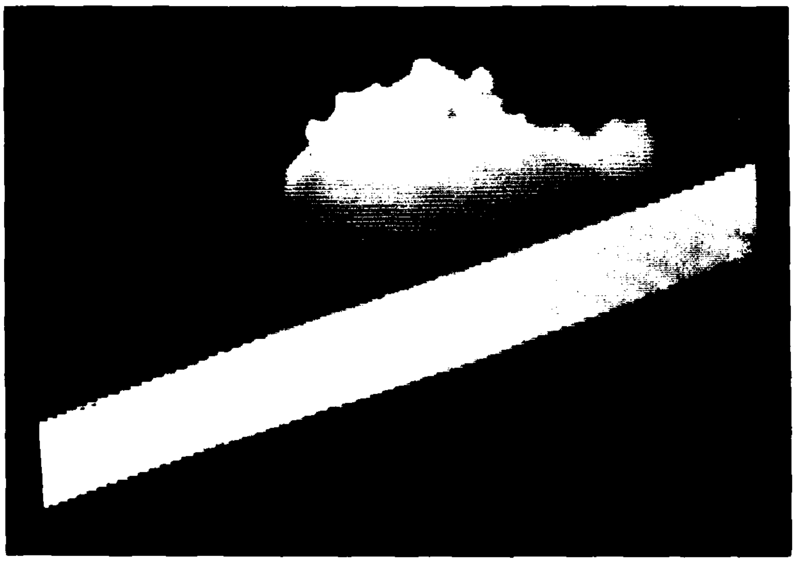

Fig. 17 A static (1/50 sec) thermal image with rotor blade influence. FLIR Systems, FLIR with 17° x 28° FOV and two detectors.

In the dynamic case the rotor blade influence is seen as a "chopper" effect occuring in the image. The eye sees by integration of the whole FOV with all the information through the blades. Rotor blade interference can produce problems for a tracker with poor filtering. However, for acquisition purposes, the LOS of the gunner's FLIR, with MFOV and NFOV, will generally be directed over the rotor blades at targets situated 2-4 km from the hovering helicopter (see Fig. 9).

4. HIGH RESOLUTION MULTI-SENSOR PACKAGE FOR COPILOT/GUNNER TASKS

Technological developments in the near future will allow the use of a multi-sensor package with high resolution and magnification factors in modern scout/combat helicopters. This multi-sensor package can comprise the following electro-optical devices:

- DVO	(0.4 - 0.7 μm)	day	passive
- TV-channel	(0.4 - 1.1 μm)	day	
- LLLTV-channel	(0.4 - 0.9 μm)	day/night	
- FLIR	(3-5 μm, 8-14 μm)	day/night	
- LRF/D	(1.06 μm, or 9.6 and 10.6 μm)		active (day and night)
- reconnaissance radar	(35, 94, 140 or 230 GHz)		
- obstacle warning system based on laser or radar techniques	(1.06 μm, 10.6 μm, 60 GHz ...)		
- Laser and radar warning system			passive

The DVO and IR sensors are normally installed on a two stage stabilized platform with a stabilization accuracy of approx. 5 - 50 μrad (1 σ), to reduce the effect of helicopter vibrations and produce good image resolution.

To better understand the physical aspects of the multi-sensor package, the relationship of sensors to wavelength will be discussed. Fig. 18 shows the attenuation of the atmospheric gases in dB/km as a function of the frequency (wavelength). The diagram contains three curves for precipitation (drizzle of 0.25 mm/h, heavy rain of 25 mm/h and excessive rain of 150 mm/h) and a curve for strong fog with 50 m visibility in the

0.4 - 0.7 µm range (ref. 10, 11 and 12). These curves have to be added to the attenuation of the gases. We are normally familiar in IR optics with the inverse curve - the atmospheric transmission - of the 0 - 15 µm wavelength range, which is shown in Fig. 19; pay attention to the different scales: linear, log. and log. log. Fig. 19 contains three further curves of interest for thermal imaging with the black body radiation (Planck's Law) for different temperatures, the IR optics transmission and the IR detector detectivity.

Fig. 18 shows clearly, why in particular radar and, to a lesser extent, a 10 µm laser and a 10 µm FLIR can penetrate dense fog with 50 m visibility better than a DVO or a TV channel. The physical effects of different attenuation as a function of the wavelength can be explaned by the Mie and Rayleigh absorption and scattering. The attenuation is strongly dependent on the particle size in the atmosphere (aerosols, molecules etc.). Rain has a much larger particle size than fog, therefore rain also affects the transmission in the IR band up to a wavelength of approx. 3 mm (100 GHz). Only excessive and heavy rain additionally influence the previously mentioned 35 and 94 GHz frequency bands. The attenuation for fog is mainly caused by absorption and not by scattering effects and is dependent on the water content and not on the particle size distribution.

The attenuation for snow and hail is much less than that for rain in the millimeter wave region and depends on the crystal formation and water content. The reason is the difference in dielectric properties between the two phases, solid and liquid. If the snow or hail is melting, the attenuation will be higher than for rain because the fractional volume is higher than for a rain drop with the same water content. This brief discussion attempts to demonstrate the usefulness of a multi-sensor package for a helicopter in adverse weather conditions.

Radar has a better penetration through fog and rain than devices operating in the IR and visible bands. A disadvantage of micro-wave or a millimeter wave radar is the reduced resolution. The diffraction limited angular resolution ϕ of a sensor is directly proportional to the wavelength λ and inversely proportional to the entrance pupil (EP) D.

$$\phi = 1.22 \quad \lambda / D \sim \lambda / D$$

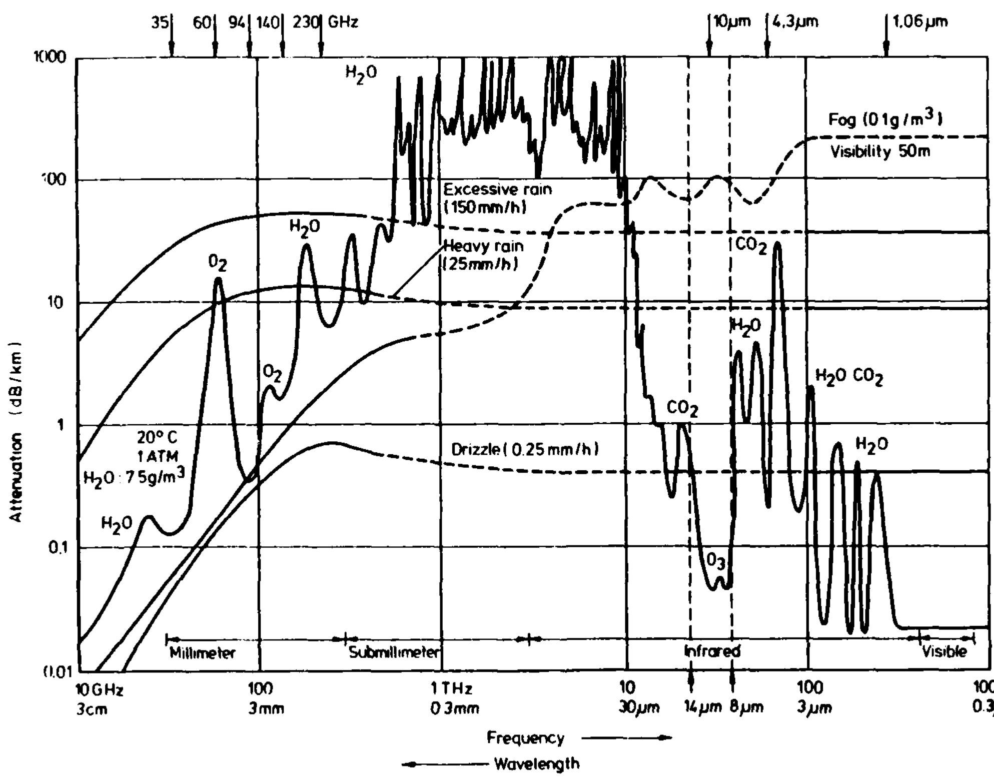

Fig. 18 The attenuation of the atmospheric gases as a function of the frequency (wavelength). Parametric attenuation curves of drizzle, heavy rain, excessive rain and strong fog are shown (ref. 10 and 11)

It is possible to detect the enemy with an active 94 GHz reconnaissance radar at a distance of 4-6 km, but an identification is not possible on the display. If a synthetic aperture radar (SAR) antenna is installed on the rotor blade, the diameter D can be increased to 10 - 20 m, which means that, with a 3 mm radar, the resolution approaches the region of FLIR resolution $\phi \approx 0.2$ mrad, but only in AZ. A conventional radar installed in a mast-mounted sight configuration has a much coarser resolution, because of the smaller aperture D.

The same considerations are valid for a wire detection system based on laser or radar principles. With a pulsed laser warning system it is possible to produce a quasi-picture of the scene. A pulsed radar warning system has the advantage of better transmission through fog and rain. The german company AEG is presently developing (ref. paper no. 43) an obstacle warning system (LPI radar) which uses the 60 and 66 GHz frequency band with the O_2 gas attenuation to reduce the detectability as an active device.

To comply with the increasing stand-off ranges of modern weapon systems the DVO, TV and IR sensors installed on a stabilized platform have a tendency to larger apertures and greater magnifications which correspond to a longer focal length. As an example the FLIR sensor will be discussed in more detail. For the visual aids several reconnaissance criteria are used for discrimination

- target acquisition
- detection (additionally hot spot detection, classification and orientation)
- recognition
- identification and
- tracking.

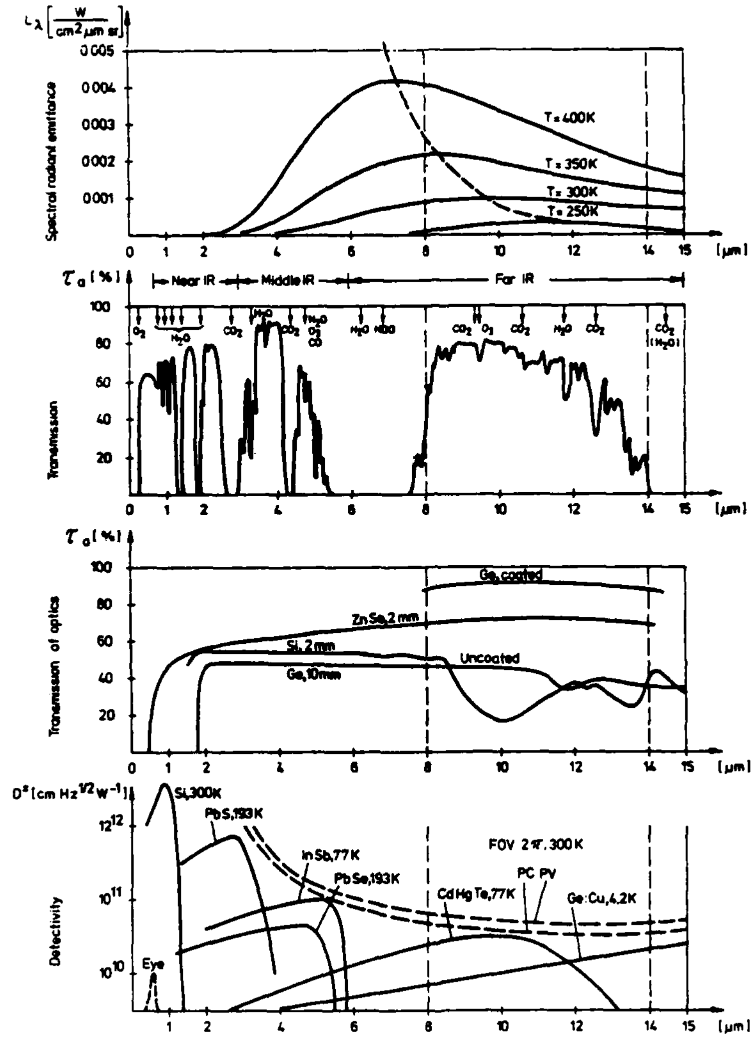

Fig. 19 Various FLIR parameters as functions of the wavelength

a) black body radiation for different temperatures (Planck's Law)
b) atmospheric transmission
c) IR optics transmission
d) IR detector detectivity with BLIP conditions

The thermal and geometric resolution of a FLIR are calculable using a model with the minimum resolvable temperature difference (MRT) formula, where it is necessary to define many FLIR parameters (ref. 13). The temperature difference ΔT is plotted against the spatial frequency (cycles per mrad). The spatial frequency can be transformed into a range with the appropriate detection, recognition or identification criteria (1 cy/mrad, 3.5 cy/mrad, 7 cy/mrad). The performance of an optical sight should cover the max. range of the weapons used. It is easy to understand that the relationship between resolution or range and FOV size should be optimized. If the FOV of the FLIR is small, providing greater range or higher spatial frequency, the observer may not be able to detect or recognize a target, because of the long search process. A high resolution FLIR should have a minimum of three FOV i.e. WFOV (detection), MFOV (detection and recognition) and NFOV (recognition and identification). A 4th FOV with $\sim 1°$ is desirable for identification but this NFOV is not good for tracking and aiming. A gunner's FLIR has the same high requirement for thermal resolution as a pilot's FLIR i.e. $\Delta T \lesssim 0.05$ K. This high resolution is necessary as enemy tanks or helicopters in future will have IR suppression to aid camouflage. Hot spot with cue identification will then no longer be possible and only visual identification will be practicable. Figs. 20 and 21 show two thermal images obtained with different FOV and FLIRs.

A properly equipped combat or scout helicopter needs LRF and a laser and a radar warning system with sensors mounted in various positions on the airframe.

Fig. 20 Thermal image ($\sim 0.5° \times 0.3°$) with the MIRA-FLIR, with addition of a 4x telescope and a TV camera for recording the LED output. A tank is traversing a wood at a distance of 3.2 km.

Fig. 21 Thermal image ($1.8° \times 2.7°$) with the CALIPSO (SMT) FLIR in the OPHELIA MMS. A group of tanks at a distance of approx. 1300 m under bad weather conditions.

5. CONCLUSIONS

The scenario of a modern battlefield encompasses so many aspects that no clear statement for either an NMS, an RMS or an MMS can be given at present. Experiments, flight tests and calculations based on vulnerability (survivability) have shown that an MMS is a good system for tactical missions with deployment from behind cover in an air-to-ground mission. The visible, IR and radar signatures of an MMS behind cover are less than those of either an NMS or RMS on account of reduced cross-section. The height of the tail rotor however also affects the chances of detection by radar (Doppler effect) of a helicopter in the MMS configuration. In an open air-to-air attack the advantages of an MMS related to the vulnerability are not significant. If a helicopter equipped with MMS has not been aerodynamically optimized, the max. speed can be degraded compared to either NMS or RMS configuration.

A high resolution multi-sensor package comprising DVO, LLLTV channel, FLIR, LRF and radar for reconnaissance purposes is necessary for the co-pilot/gunner to provide a combat helicopter with 24 hour capability under virtually all weather conditions. A reconnaissance radar has the advantage of being able to successfully survey the battlefield even in bad weather. The disadvantages of this sensor are that it is an active system and has lower resolution compared to a DVO or a FLIR, if an SAR in the rotor tips is not used. A high performance FLIR will solve many observation problems in either day or night battles.

The weapon systems have to be matched to the high resolution visionics system and vice versa.

6. ABBREVIATIONS

AFCS	Automatic Flight Control System
AGL	Above Ground Level
ANVIS	Aviators Night Vision Imaging System
AZ	Azimuth
BLIP	Background-Limited Infrared Photodetection
CALIPSO	CAméra Légère Infra-rouge Pour Systeme OPHÉLIA
CCIR	European video standard with 625 lines, 25 Hz frame rate
CM	Common Modules
CEP	Circular Error Probability
CRT	Cathode Ray Tube
DVO	Direct View Optics
EIA	American video standard with 525 (875) lines, 30 Hz frame rate
El	Elevation
EP	Entrance Pupil
FLAB	Flying Laboratory
FLIR	Forward Looking Infrared
FOV	Field Of View
HDD	Head-Down Display
HMD	Helmet-Mounted Display
HMS	Helmet-Mounted Sight
HMS/D	Helmet-Mounted Sight/Display
HUD	Head-Up-Display
IAS	Indicated Air Speed
IFOV	Instantaneous Field of View
IR	Infrared
ISA	International Standard Atmosphere
KIAS	Knots Indicated Air Speed
LASER	Light Amplification by Stimulated Emission of Radiation
LLLTV	Low Light Level TV camera
LOS	Line Of Sight
LPI	Low Probability of Intercept
LRF/D	Laser Range Finder/Designator
MCP	Maximum Continuous Power
MFOV	Medium Field Of View
MIRA	MILAN Infrarot Adapter
MMS	Mast-Mounted Sight
MRT	Minimum Resolvable Temperature Difference
NMS	Nose-Mounted Sight
NFOV	Narrow Field Of View
NOE	Nap of Earth
NVG	Night Vision Goggles
OPHÉLIA	Optique sur Plate-forme HELIcoptère Allemand
PAH	Panzer Abwehr Hubschrauber (tank defense helicopter)
PC	Photoconductive detectors
PISA	Piloten Infrarot Sicht-Anlage (Pilots infrared system)
PNVS	Pilot Night Vision Sensor
PV	Photovoltaic detectors
PVS	Pilot Vision System
RADAR	Radio Detection and Ranging
RMS	Roof Mounted Sight
SAR	Synthetic Aperture Radar
SG	Symbol Generator
SPRITE	Signal PRocessing in The Element (TED)
TADS	Target Acquisition Designation Sight
TED	Tom Elliot Device (SPRITE)
WFOV	Wide Field Of View

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NOUVELLES METHODES D'ETALONNAGE DE L'ANEMOMETRIE DES HELICOPTERES AUX BASSES VITESSES

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RESUME

Le but de cet exposé est de présenter les travaux d'améliorations réalisés par Crouzet sur les méthodes classiques d'étalonnage d'anémométrie pour les rendre compatibles avec le domaine de vol des hélicoptères et les nouveaux produits proposés. On examine les limitations des méthodes classiques, les améliorations définies et on présente le matériel mis en place pour ces nouvelles méthodes au Centre d'Essais en Vol de Brétigny.

1 - INTRODUCTION

Les systèmes anémométriques des prochains hélicoptères armés devront être capables de fournir des informations fiables et précises sur le vecteur vitesse air dans tout le domaine de vol, en particulier, dans le domaine des basses vitesses, c'est à dire de module inférieur à 20 m/s, et d'orientation totalement quelconque.

Les produits proposés par Crouzet répondent à ce besoin critique :

- pour assurer la sécurité du pilotage dans des conditions extrèmes ; de nuit, par mauvais temps, en mer ou en montagne,

- pour les systèmes de conduite de tir. Une bonne précision est particulièrement nécessaire pour les armes non guidées après le tir telles que les canons ou les roquettes.

Crouzet étudie avec le soutien du Service Technique des Télécommunications et Equipements Aéronautiques (STTE) du Ministère Français de la Défense la définition de nouveaux principes de mesure de la vitesse des hélicoptères :

- les méthodes de détermination de la vitesse-air par moyens internes (concept VIMI, en liaison avec l'inventeur et l'Aérospatiale)

- l'analyse en vol du champ aérodynamique de l'hélicoptère afin d'y qualifier les zones propices à des mesures anémobarométriques. Pour celà, une sonde a été développée spécialement par Crouzet

- l'anémométrie laser.

Cependant, la mise en oeuvre des installations anémométriques aux basses vitesses suppose au préalable la possibilité de faire des étalonnages précis. C'est pour celà qu'un des volets de l'étude précédemment citée concerne, en liaison avec le centre d'essais en vol (CEV) de Brétigny, l'amélioration des méthodes d'étalonnage. Plus particulièrement :

- les spécifications des moyens de référence anémométriques nécessaires à l'évaluation de futurs systèmes anémométriques

- les limites et les performances des moyens actuels

- les améliorations possibles et les nouveaux principes afin de tenir les objectifs définis.

2 - SPECIFICATION DES PERFORMANCES NECESSAIRES

Les performances des systèmes de conduite de tir déjà mentionnés conduisent à envisager une précision d'étalonnage de l'ordre du noeud, sur chacune des composantes long et travers de la vitesse air (Vx et Vy). Le domaine de mesure couvre la totalité du domaine de vitesse des hélicoptères, soit approximativement - 20 m/s à + 100 m/s en vitesse d'avancement et - 20 à + 20 m/s en vitesse latérale. La bande passante utile reste modérée, typiquement du continu à deux Hertz.

3 - MESURE DU VENT AU VOISINAGE DU SOL

Quelle que soit la méthode d'étalonnage, il est nécessaire de connaître les variations spatio-temporelles du vecteur vitesse du vent. De ce fait Crouzet a entrepris une analyse statistique, menée essentiellement expérimentalement à une hauteur voisine de 10 mètres où se font principalement les essais d'étalonnage.

Ces essais sur deux sites ont permis de compléter le peu d'informations disponibles à ce jour sur la nature du vent en site dégagé caractéristique des centres d'essais en vol :

- l'aérodrome de Valence Chabeuil, dont la piste est orientée Nord Sud dans le sens des vents dominants de la vallée de Rhône.

- l'aérodrome de Brétigny, à proximité de la piste principale d'essais en vol orientée Nord/Est - Sud/Ouest (planche 1).

Dans les deux cas, ces mesures ont été effectuées en haut d'un mât à l'aide d'un tube de pitot monté sur une girouette s'alignant librement dans le vent. A partir de l'écart entre la pression d'arrêt et la pression statique ambiante, de la température statique et de la position de la girouette en azimut, on obtient simplement la vitesse du vent et sa direction.

Des analyses spectrales faites en laboratoire sur le matériel réel avaient montré que la bande passante de l'installation complète (tube de pitot, canalisations, micromanomètre) était de l'ordre de 4,5 Hz pour le module du vent. Les résultats apparaissent sur la planche 2 pour trois types de vents : faibles (inférieurs à 2 m/s), moyens (2 à 5 m/s) et forts (supérieurs à 5 m/s). L'allure des spectres est identique pour tous les vents et on constate que :

- lorsque la fréquence augmente, l'énergie des variations de l'intensité du vent décroît avec une pente moyenne identique (- 17 dB par décade). Ce résultat est confirmé par des essais de la Royal Meteorological Society (cités dans la référence 1).

- l'énergie des variations de la direction du vent diminue à partir d'une fréquence qui est d'autant plus basse que le vent est faible.

Par ailleurs, on montre que les variations d'intensité ne sont pas corrélées aux variations de vitesse, et surtout que les écarts types en intensité augmentent notablement avec le vent moyen.

Cet important résultat est confirmé théoriquement. Moyennant les hypothèses d'homogénéité de la couche limite superficielle, établie et stable, on peut établir la relation suivante :

$\sigma u \simeq \bar{u}/\mathrm{Log}\,\frac{Z}{Z_0}$ (1) où σu représente l'écart type de la fluctuation du module de la vitesse du vent à la hauteur Z, Z_0 l'indice de rugosité caractéristique du terrain et $\bar{u}$ le vent moyen (référence 2). Pour une zone dégagée l'indice Z_0 varie de 2 millimètres pour l'herbe rase à 100 pour des buissons ou arbustes.

Cette relation (1) permet, après détermination expérimentale de l'indice de rugosité soit de déterminer le vent maximal à une altitude donnée, soit l'altitude minimale pour un vent donné où l'on peut effectuer des essais en vol significatifs.

4 - METHODES CLASSIQUES

4-1 Méthode par mesure locale du vent

Cette méthode consiste à utiliser la relation $\vec{V_a} = \vec{V_S} + \vec{V_W}$ où $\vec{V_S}$ est le vecteur vitesse sol instantané, $\vec{V_W}$ le vecteur vent instantané et $\vec{V_a}$ le vecteur vitesse-air.

Les mesures de vitesse-sol peuvent être précises et fiables grâce à une instrumentation au sol (radar, Lidar) ou embarqué (radar Doppler, centrale inertielle recalée fréquemment, etc...).

Le mesure de vitesse du vent peut être faite grâce à un anémomètre monté sur un mât à une hauteur voisine de la hauteur de vol de l'hélicoptère (Planche 3). L'hélicoptère doit voler à vitesse sol stabilisée et passer à proximité de l'anémomètre.

Il est clair que lorsque l'aéronef sera près du mât, une perturbation importante sera apportée par le souffle du rotor et la mesure sera inexploitable. Par contre, s'il passe loin de manière à minimiser l'interaction, les mesures du vent risquent de ne plus être significatives. D'où la nécessité d'utiliser les valeurs du vent moyen autour du temps de passage de l'hélicoptère au voisinage du mât, l'erreur sur l'estimation de la vitesse du vent étant de ce fait fonction de l'écart-type de ses variations.

Il apparait, aux très faibles vitesses de vent que la hauteur de vol a une influence modérée autour de 10 m compte tenu des erreurs existant par ailleurs. L'écart-type de l'erreur croît globalement avec la vitesse moyenne du vent. Ces résultats sont représentés à titre d'exemple planche 4. La mise en oeuvre de cette méthode n'est donc envisageable que par vents modérés (typiquement inférieurs à 2 m/s) et de plus nécessite de nombreux passages stabilisés au voisinage d'un anémomètre.

4-2 Méthode d'aller-retour

Il s'agit d'effectuer des allers-retours à vitesse air constante au dessus d'une base de longueur L matérialisée au sol. Si l'on suppose que le vent est constant lors du survol aller de durée T_1 et du retour de durée T_2, la vitesse air s'exprime simplement par :

$$V_a = \frac{L}{2} \left(1/T_1 + 1/T_2 \right) \qquad (2)$$

Il n'y a donc pas nécessité de mesurer le vent.

Pour fixer les ordres de grandeur, si on suppose que le vent varie de 1 m/s entre l'aller et le retour - ce qui est une hypothèse simpliste - on montre que l'estimation de la vitesse-air par la formule (2) est entachée d'une erreur de 0,5 m/s.

L'erreur totale (somme quadratique de l'erreur précédente et des erreurs de mesure) croît avec la vitesse air, de manière d'autant plus importante que la base est courte.

Pour une base de longueur donnée, les erreurs de mesure ont une influence d'autant plus grande que la vitesse-air est grande et l'écart du vent aller-retour est petit (planche 5). L'inconvénient majeur de cette méthode est que l'erreur dominante dans la gamme de vitesse-air considérée est liée à la stabilité du vent.

4-3 Amélioration de ces méthodes

Une solution consiste à mesurer la vitesse du vent en différents points alignés matérialisant une base au sol. La mesure de la longueur de cette base peut être soit physique, soit déduite de l'intégration a postériori de la vitesse-sol mesurée à bord de l'hélicoptère.

Elle s'apparente à la première méthode mais au lieu de répéter plusieurs fois la même passe pour moyenner les effets du vent, on peut directement moyenner les informations pendant un certain temps, se rapprochant ainsi de la méthode du passage sur base.

Pour une base de longueur L, parcourue en un temps t_1 à une vitesse-air V_a constante, on a :

$$V_a = \frac{L}{t_1} - \overline{V_W}^{T} - \frac{1}{t_1} \int_0^{t_1} V_W' \, dt, \text{ si } V_W' \text{ est la partie à moyenne nulle du vent.}$$

On peut écrire, en effet à chaque instant que $V_W(t) = \overline{V_W}^T + V_W'(t)$, la moyenne $\overline{V_W}^T$ du vent étant prise sur un temps T a priori supérieur à t_1, par exemple en enregistrant la valeur du vent avant et après les passes. L'estimation du temps T nécessaire sera faite lors de la campagne d'essais qui vient de commencer.

Le terme $\frac{1}{t_1} \int_0^{t_1} V'_W \, dt$ est un terme d'erreur dont on peut estimer l'écart-type, thêroriquement à partir de l'allure spectrale du vent mesurée au sol pendant l'essai, compte tenu du fait qu'il est voisin de celui du vent à chaque instant au voisinage de l'hélicoptère.

Finalement, on peut obtenir une estimée de l'écart-type global à partir des écart-types sur les mesures de longueur, de temps, de vitesse du vent et des variations du vent (planche 6).

L'écart-type global croît avec la vitesse, d'autant plus vite que la base est courte.

A longueur de base donnée, l'erreur croît avec le vent moyen. Cependant, on obtient des valeurs compatibles aves les objectifs visés pour des vents modérés (jusqu'à 8 m/s) dans le domaine de vitesses considéré.

5 - METHODE DITE DU VEHICULE SUIVEUR

On vient de voir l'importance des fluctuations du vent et de leur prise en compte. C'est pourquoi cette méthode repose sur la mesure continue du vent. L'hélicoptère est astreint à suivre un véhicule équipé d'une anémomètre précis monté sur un mât télescopique.

Le principe de l'expérience est décrit sur la planche 7. Le véhicule et l'aéronef sont munis de dispositifs de mesure de vitesse-sol. La vitesse du vent est déduite de la mesure anémométrique et de la vitesse sol du véhicule et est additionnée vectoriellement à la mesure de la vitesse sol de l'hélicoptère en tenant compte d'un léger décalage temporel.

Une démarche voisine consiste à égaler vitesse-air de l'hélicoptère et vitesse de l'anémomètre.

6 - PROGRAMME D'ESSAIS

Le but des essais est de valider et de comparer les méthodes d'étalonnage en vol afin d'en qualifier au moins une.

L'hélicoptère utilisé pour les essais est équipé d'une référence de vitesse sol performante, centrale inertielle à plateforme ou radar doppler.

Pour obtenir une référence de vent, le CEV met en oeuvre une base d'une longueur d'environ 1200 mètres de longueur, utilisée sous les deux aspects cités plus haut :

- passage sur base munie d'anémomètres
- suivi d'un véhicule muni d'un anémomètre et d'une cinquième roue tachymétrique.

6-1 Choix du site

L'implantation de la base d'étalonnage sur la plateforme d'essais de Brétigny sur Orge a été choisie en fonction de divers éléments liés aux conditions atmosphériques locales, à l'éloignement des obstacles perturbant l'aérologie.

D'après des statistiques conduites sur les 25 dernières années, les vents dominants sont du 220 puis du 020, l'heure la plus calme étant 2 HTU, la plus ventée 15 HTU, le mois le plus calme octobre et le plus venté avril.

Selon ces contraintes, la base retenue est le taxiway de la piste principale, orienté au 35-215, recouvert d'un excellent revêtement permettant d'éviter tout à-coup sur le mat.

6-2 Mesure du vent aux points fixes

* L'installation météorologique classique disponible sur le terrain de Brétigny, bien que robuste et facile à mettre en oeuvre, est insuffisante : bandes passantes de l'ordre du Hertz et précisions insuffisantes en direction et vitesse seulement dans le plan horizontal. Ces informations ne seront utilisées que qualitativement.

* Deux pylones de 8 mètres situés à proximité de la base d'étalonnage sont équipés chacun d'un même moyen anémométrique : sonde pitot statique montée sur une girouette associée à un capteur différentiel et à une sonde de température.

* Les composantes du vent sont mesurées uniquement dans le plan horizontal.

Les paramètres temps universel, vitesse et direction du vent, température de l'air sont élaborés et numérisés par des unités centrales modulaires BOA et transmises sur des lignes souterraines jusqu'à la station Charlemagne pour enregistrement à 32 Hz sur bandes magnétiques.

6-3 Mesure du vent sur les différents points de la base (planche 9)

* Le véhicule CARAMEL (camion radio mesures élaborées) utilisé est une fourgonnette Renault Trafic, équipée d'un mât pneumatique télescopique de 10 mètres qui supporte une sonde tridimensionnelle à films chauds associée à son électronique de mesure. Une campagne d'essais préliminaire d'abord sur véhicule léger puis sur le véhicule CARAMEL a permis de vérifier l'absence de couplages entre les vibrations et la mesure anémométrique aux basses fréquences qui pouvaient compromettre la qualité de la mesure. L'étalonnage de la sonde a lieu en soufflerie.

* Les paramètres sont enregistrés sous forme analogique. La bande magnétique est ensuite numérisée à 32 Hz et contient les paramètres suivants :

temps universel

3 composantes de vitesse air en axe sonde

vitesse sol (5° roue tachymétrique)

3 lectures accélérométriques au niveau de la sonde.

La synchronisation est faite par UHF sur l'horloge maîtresse.

6-4 Mesure à bord de l'hélicoptère (planche 3)

* L'hélicoptère utilisé est le SA 330 Puma n° 1024. Son autonomie est de 2 heures, sa charge utile pour les matériels en essais de 1 tonne et sa masse maximale au décollage de 9 tonnes.

* Sur ce Puma sont avionnés en référence une centrale inertielle à plateforme (informatiquement adaptée pour les essais en vol) un radar doppler, un anémomètre compensé Crouzet type 51.

. La précision en vitesse sol de la référence inertielle (évaluée lors de sa réception sur la machine) est de 0,6 KT à 1σ par axe et, après correction des erreurs en temps différé par la "méthode des posés" de 0,1 KT à 1σ par axe.

. Le doppler fournit ses informations avec les précisions suivantes :

0,5 % ou 0,25 KT en vitesse horizontale

0,5 % ou 0,8 KT en vitesse verticale

. L'anémomètre 51 donne une vitesse propre meilleure que 2,5 KT à 3σ au dessus de 50 KT.

* L'installation de mesures se compose essentiellement d'un calculateur 16 bits, d'une unité d'acquisition mixte, d'un enregistreur numérique fournissant des bandes magnétiques, d'un enregistreur numérique à cassettes, de deux écrans de visualisation cathodiques 5" et des divers coupleurs permettant le dialogue entre ces systèmes.

Le logiciel du calculateur assure les fonctions d'acquisition, de prétraitement vers l'enregistreur et de traitement en temps réel des paramètres, affichés en grandeurs physiques sur les tubes cathodiques.

* Les bandes magnétiques, transcodées comportent à raison de 2000 mots par seconde :

le temps universel

les paramètres de référence inertiels (cap, attitudes vitesses en axes machine et géographique, vitesses de rotation et accélérations à 25 Hz).

les trois composantes de la vitesse doppler

l'incidence et le dérapage à 8,33 Hz

et les données des systèmes anémométriques en essai.

La synchronisation s'effectue par voie UHF sur émission de l'horloge maîtresse. Compte tenu de la dérive des différentes horloges embarquées, l'ensemble des paramètres (sol et Puma) est isodaté à la milliseconde près.

6-5 Conduite des essais

Les services concernés (sécurité, contrôle, pilote, etc...) ayant été prévenus à l'avance, l'axe préférentiel est choisi le matin même de l'essai en fonction du vent dominant. Après accord du contrôle pour l'utilisation du taxiway et un briefing général, l'essai peut commencer.

La synchronisation faite et vérifiée vocalement, les passages débutent. L'hélicoptère suit la voiture à une distance d'autant plus faible que la vitesse est grande pour éviter les perturbations du flux rotor et à une hauteur dépendant du domaine de vol de sécurité (typiquement 30 à 50 ft). Pour des raisons de sécurité, une deuxième voiture est utilisée pendant les essais à vitesses négatives. Une première estimation du vent lors d'un passage à vitesse moyenne permet une conduite d'essais en vitesse-air approchée, la mesure fine étant faite lors de chaque passe.

Les bandes sont alors dépouillées en temps différé au CEV et chez l'industriel.

Trois chaînes informatiques sont utilisées :

- une "temps réel" pour la conduite de l'essai

- deux en temps différé à partir des bandes magnétiques et des enregistrements en vol sur cassettes. Cette deuxième exploitation permet de valider très rapidement l'essai sur le site du CEV, les calculs complets intervenant à l'aide des données de la première chaîne.

7 - PREMIERS RESULTATS : CONCLUSION

Plusieurs vols ont déjà été effectués pour la mise au point préliminaire du matériel et de la conduite de l'essai. Le matériel est présenté planches 10 et 11. Celle-ci se fait de manière très souple et est très facile par vent modéré. L'analyse est en cours .

On peut noter que le suivi de l'hélicoptère se fait de manière parfaite et qu'on pourra peut être ultérieurement simplifier considérablement le dépouillement après analyse détaillée des résultats en assimilant la vitesse de l'hélicoptère à celle mesurée par l'anémomètre du camion CARAMEL. On disposera donc d'une méthode qualifiée d'étalonnage de la vitesse-air des hélicoptères utilisable pour les systèmes anémométriques des hélicoptères futurs.

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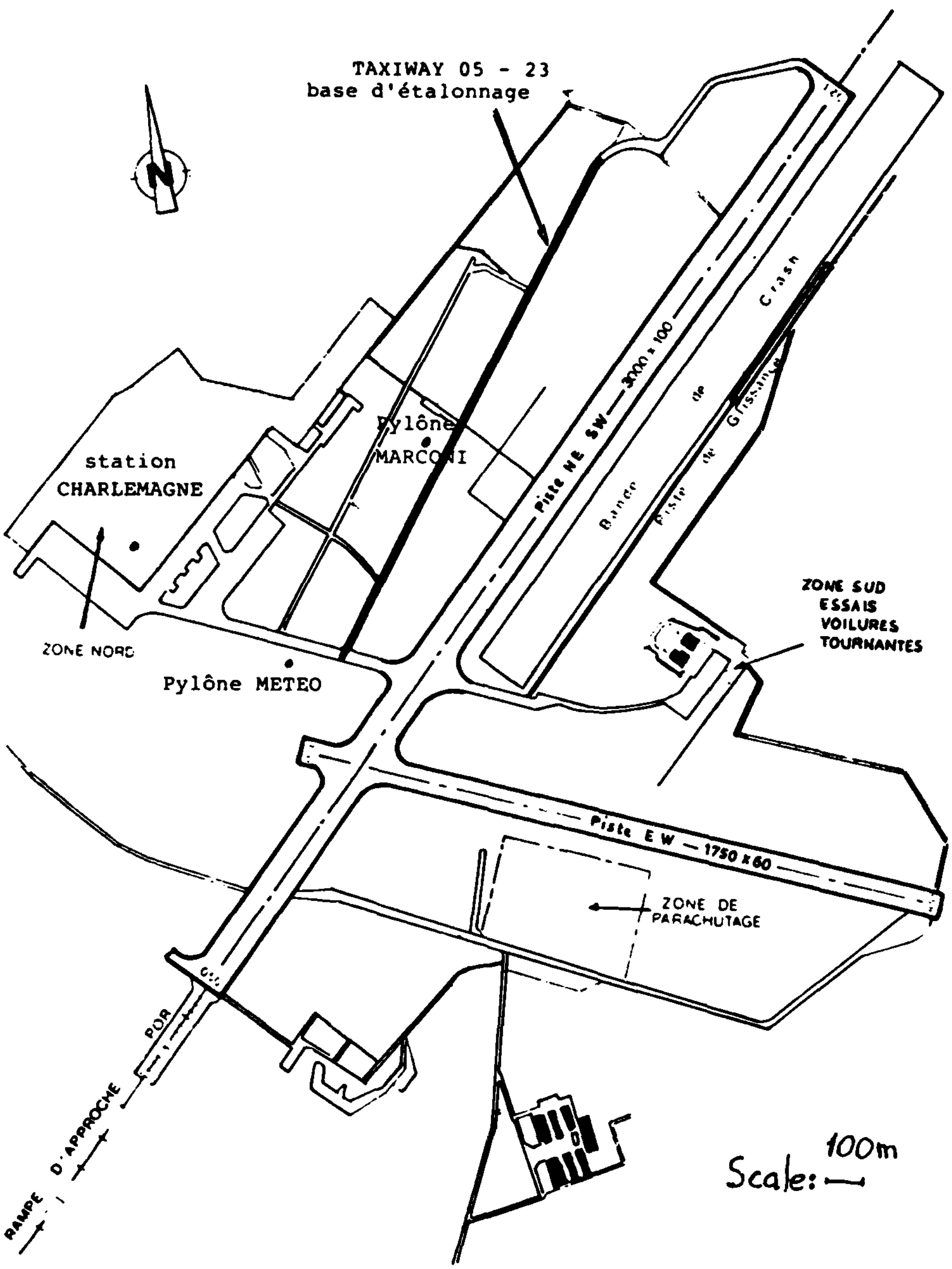


PLANCHE 1 : BASE D'ESSAIS DE BRETIGNY

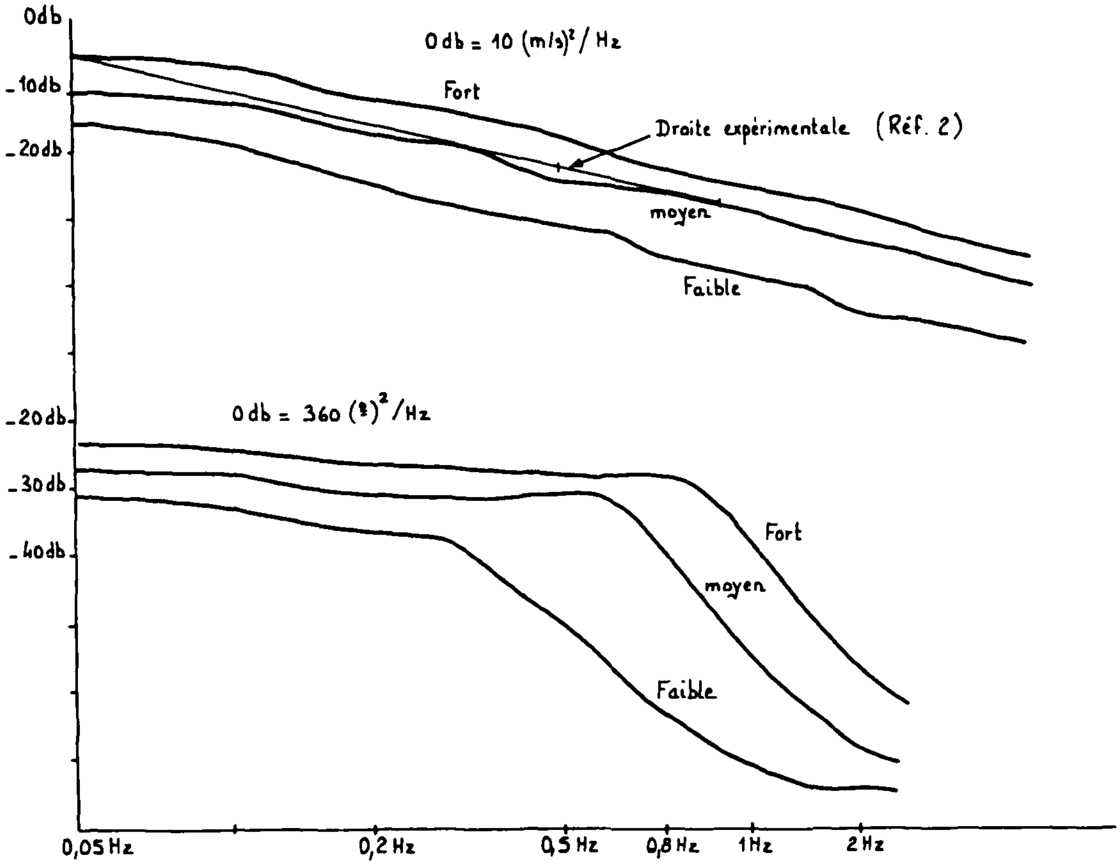


PLANCHE 2 : SPECTRES DE PUISSANCE DES VARIATIONS DU VENT EN MODULE ET DIRECTION

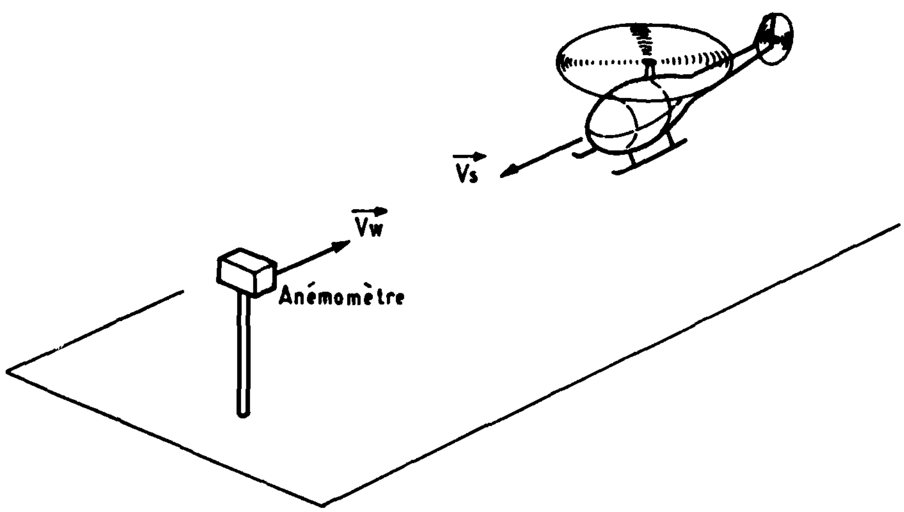


PLANCHE 3 : PRINCIPE DE LA METHODE DIRECTE

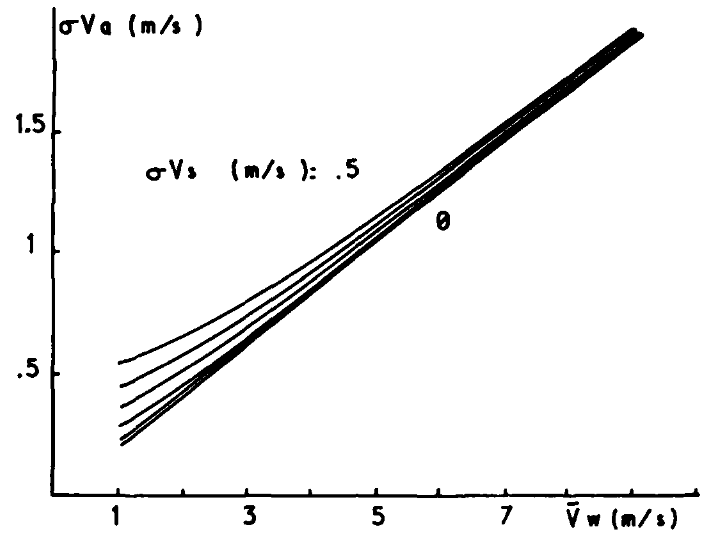
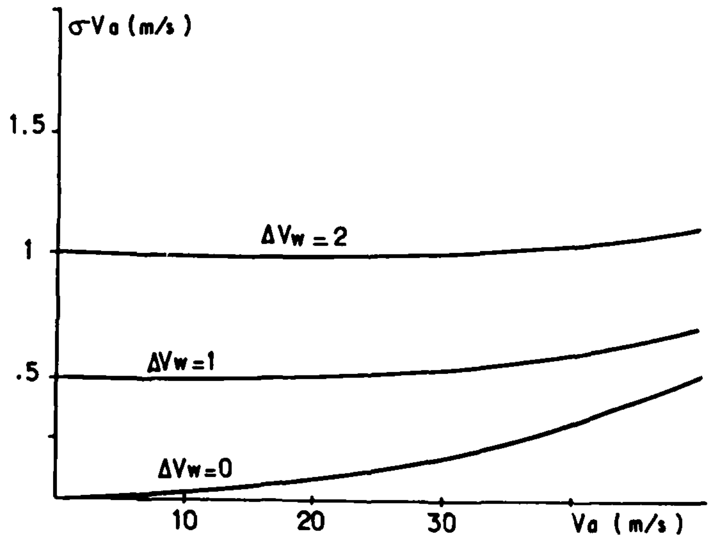


PLANCHE 4 : INFLUENCE DE L'ERREUR DE VITESSE SOL SUR LA PRECISION DE LA MESURE DE VITESSE AIR

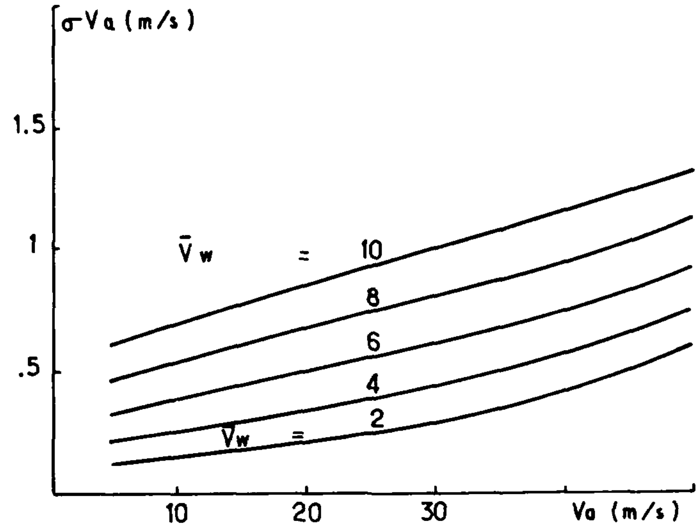


Longueur de la base : 1000 m ± 1 m

Erreur sur le temps : 0,2 s

Variation du vent : V_W (m/s)

PLANCHE 5 : ECART-TYPE DE L'ERREUR TOTALE LORS DU PASSAGE SUR BASE ALLER-RETOUR



Longueur de base : 1000 m
Hauteur : 10 m
Rugosité : 8 mm
Ecart type de l'erreur - sur le temps : 0,2 s
- sur la longueur : 1 m
- sur le vent moyen : 0,1 m/s
Vent moyen : V_W (m/s)

PLANCHE 6 : ERREUR GLOBALE EN FONCTION DU VENT MOYEN ET DE LA VITESSE AIR

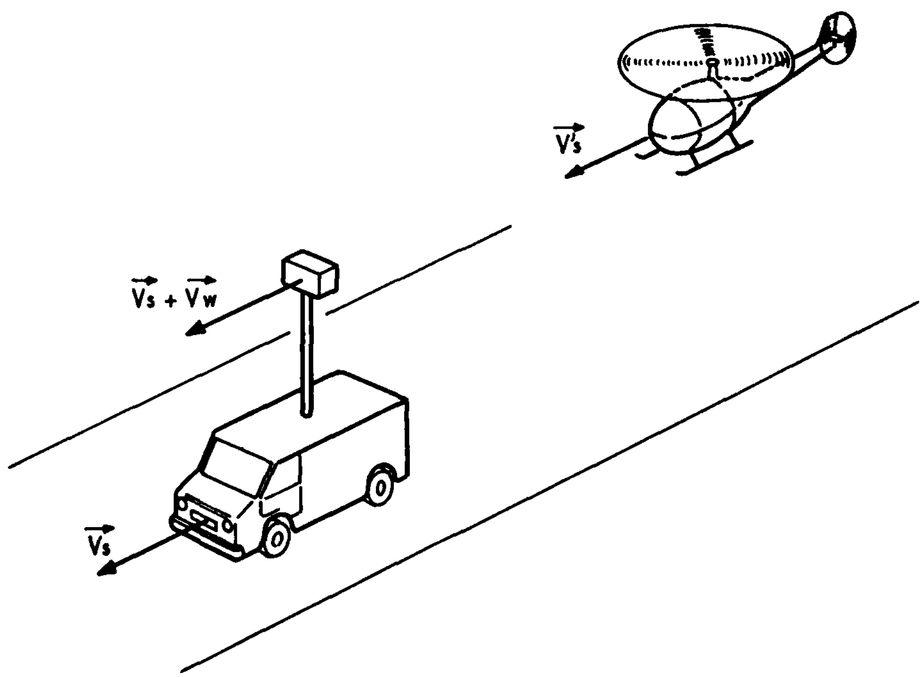


PLANCHE 7 : PRINCIPE DE LA METHODE DU VEHICULE SUIVEUR

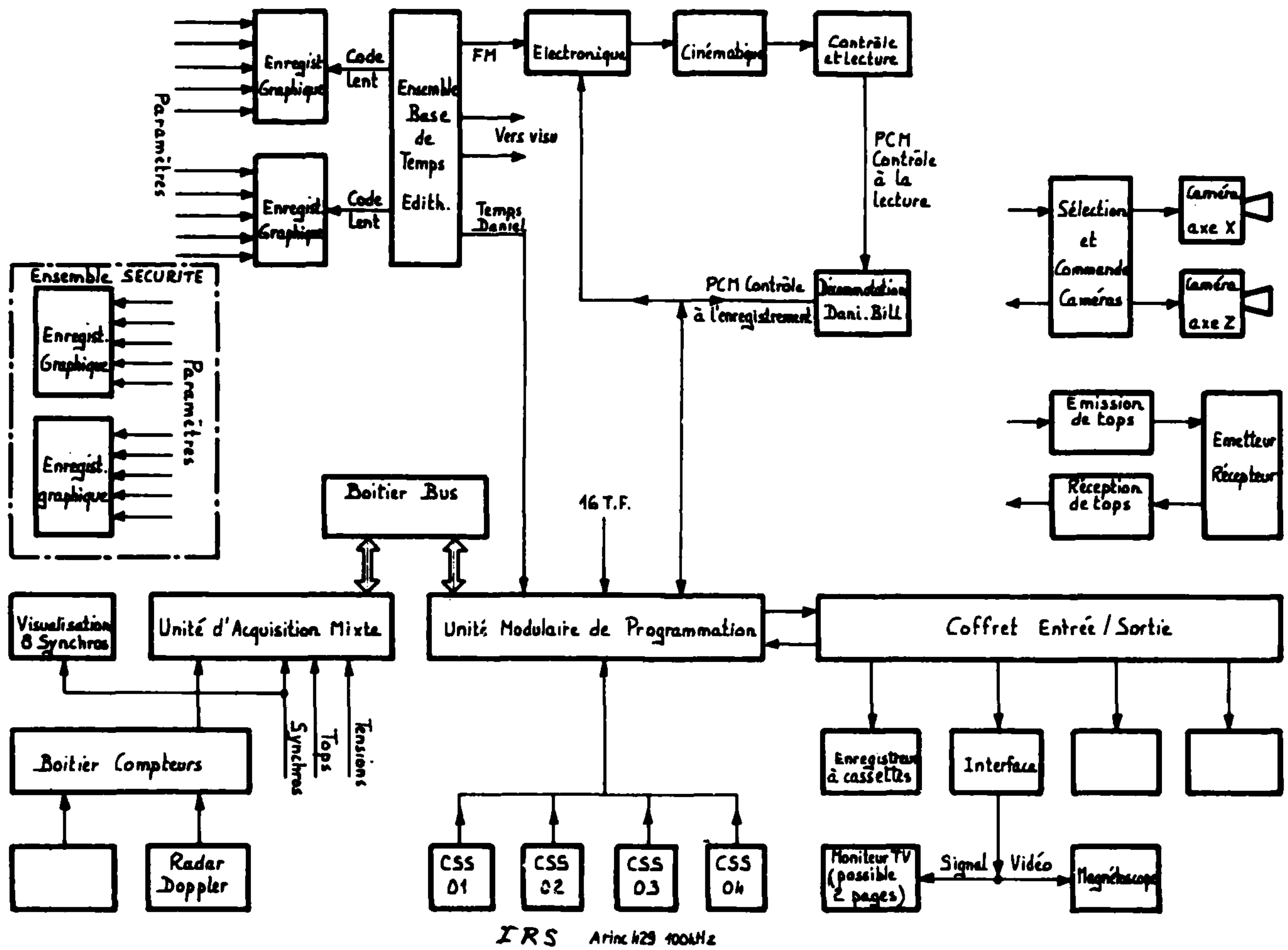


PLANCHE 8 : SYNOPTIQUE DE L'INSTALLATION EMBARQUEE

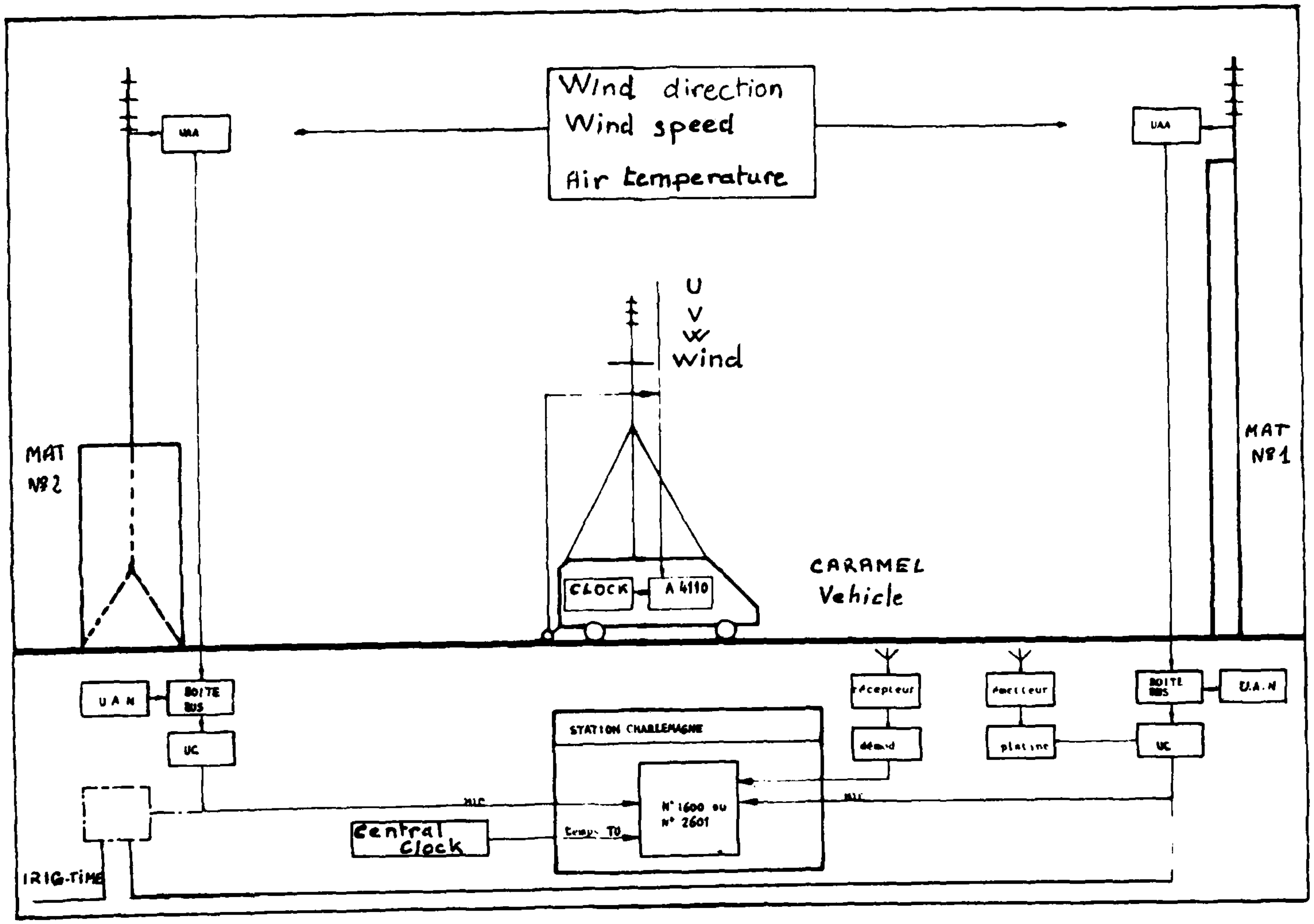


PLANCHE 9 : SYNOPTIQUE DE L'INSTALLATION DE MESURE SOL

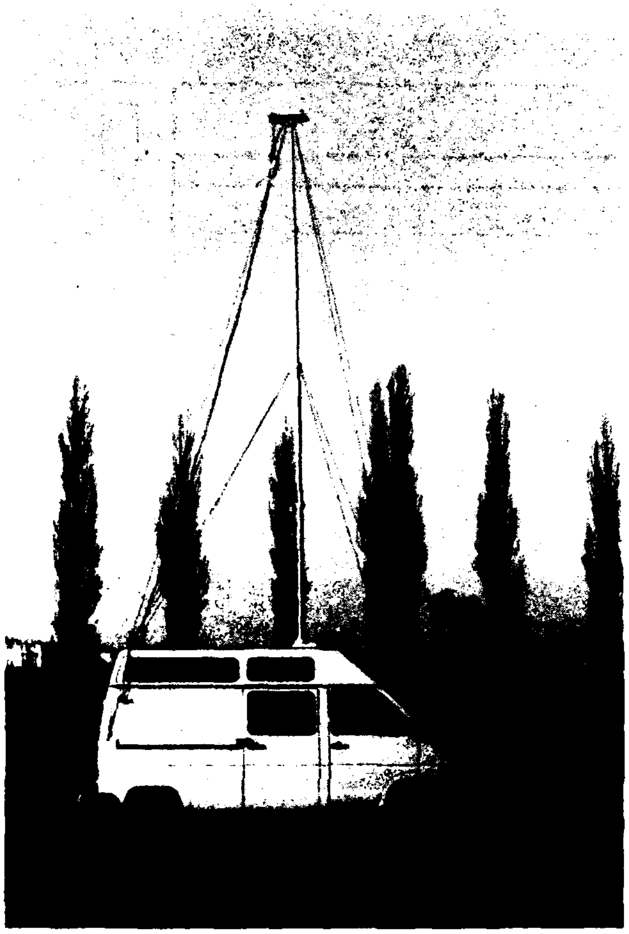


PLANCHE 10 : VEHICULE CARAMEL

PLANCHE 11 : PUMA 1024

LPI-RADAR FOR HELICOPTER OBSTACLE WARNING

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SUMMARY

Employing the frequency range of maximum atmospheric attenuation around 60 GHz a short range obstacle warning radar for helicopters has been developed, which can be operated under LPI conditions (Low Propability of Intercept). Test flights with this noncoherent solid state radar have shown that power cables as well as tall trees can be detected, even under adverse weather conditions.

System considerations and test results are described in this contribution.

INTRODUCTION

The use of helicopters for military purposes today is becoming more and more widespread. In the beginning, in the early fifties, reconaissance was the only task for military helicopters. Today transport and antiarmor engagement are standard missions; and helicopter hunting is no longer under discussion only but something for the future.

The consequent use of helicopter properties for these missions in the battlefield environment demands operation at low altitude and in unknown terrain. Only the use of natural hideouts like small hills or an accumulation of trees - i.e. NOE-flight conditions, (Nap of earth) - reduces the risk of advanced helicopter missions.

In the past helicopter accidents caused by collisons with obstacles have occured quite often. About 40% of all helicopter accidents are due to this fact as a statistic of the german Luftfahrtbundesamt for the years 1975 to 1980 /1/ has shown.

Especially dangerous are wire-like obstacles, like high-tension lines, funicular (aerial-tram) cables or antenna installations. This is due to the fact that such obstacles often cannot be seen by the pilot. Hazy weather, low standing sun or twilight reduces the optical identification drastically.

REQUIREMENTS FOR OBSTACLE WARNING

The described circumstances lead to the need of an obstacle warning sensor for helicopters. Such a sensor should fulfill the following requirements:

- obstacle detection in general
- wire detection for wire diameters as small as 3 mm
- 500 m detection range
- adverse weather capability
- small volume and weight

and especially important for military missions

- Low Propability of Intercept (LPI) - operation

Obstacle detection sensors for this purpose have been under development for several years; electromagnetic field sensors (50 or 60 Hz) /2/ and laser sensors /3/ as well as cable cutters /4/ have been investigated.

Field sensors show a high false alarm rate, especially over populated areas like cities and besides that power-lines that are turned-off or funicular cables cannot be detected.

Laser sensors have disadvantages resulting from their propagation behaviour through fog and rain or under battlefield conditions, like smoke and dust.

Cable cutters could be employed successfully, but they should be used as a last resort only.

There is an additional approach fo fulfill the above stated requirements, the employment of a mm-wave radar sensor. Compared to EO/IR systems the mm-wave approach offers better penetration capabilities, especially under adverse weather conditions.

ATMOSPHERIC PROPAGATION BEHAVIOUR

The atmospheric attenuation due to fog and rain for wavelengths from 10 cm to 0.1 μm is desplayed in fig. 1. /5/ At 300 GHz the fog attenuation amounts to slightly more than 1 dB/km, while this value increases to nearly 300 dB/km at a wavelength of 1 μm. The rain attenuation for both frequency ranges cannot be neclected, but is only of minor importance for system design.

The transmission behaviour of a 94 GHz radarsignal and an IR-signal through TOT-barrage shooting (Time On Target) is compared in fig. 2. /6/. The IR-signal is nearly blinded for more than 20 seconds, while the mm-wave signal is decreased for about 3 seconds only. The attenuation of the transmission level amounts to 15 dB (radar-signal, i.e. two way transmission).

DESIGN CONSIDERATIONS

In general the frequencies taken for mm-wave short-range radar applications are within the atmospheric windows, around 35 and 94 GHz /5/. For specific purposes however, frequencies with high atmospheric attenuation like 22, 60 or 120 GHz might be chosen. At 60 GHz for example the clear-air attenuation amounts to 16 dB/km;
to reconnoitre a source of radiation at this particular frequency will be quite diffcult. Thus a 60 GHz obstacle warning radar can be operated under LPI-conditions, the most desirable military requirement. Compared to a warning radar operating at 50 GHz for example, having a reconnaissance range of about 55 km, the 60 GHz device has a reconnaissance range of only 2.7 km under the same detection range and reconnaissance receiver conditions. What's more, separated 60 GHz radar systems operating near one another and using the same frequency do not mutually interfere.

In 1980 the first 60 GHz radar sensor was built for radar cross-section investigations of high tension line wires. Fig. 3 displays this unit.

The achieved test results /7/ demonstrated the feasibility of the 60 GHz approach to fulfill the above given requirements. Thus in a second step a scannable 60 GHz warning sensor has been developed to carry out flight tests.

TECHNICAL DESCRIPTION

A detailed description of the RF-components of this noncoherent 60 GHz pulse radar has been given in /8/. The technical data are as follows:

Transmitter frequency	59.2	GHz
Output power (peak)	4	W
Pulse width	20	ns
PRF	125	kHz
Noise figure	13.5	dB

Fig. 4 shows the radar unit mounted to the nose of a testbed helicopter.

The radar's field of view covers a wide azimuth angle: 180 degrees. In elevation, an angle of 30 degrees was chosen. With 20 ns pulses, the resolution is 3 m. The scanning mechanism to obtain these data consists of a fixed dish antenna and a movable plane mirror. The mirror rotates while swinging in its second axis from 37.5 degrees to 52.5 degrees, thus twisting the vertically incident beam between -15 degrees and + 15 degrees in elevation. The actual position in azimuth and elevation as well as the radar video signal containing the range information are fed to a micro-processor generating a C-scope display on a CRT.

The scanner was built for 450 RPM. Thus on updating frequency of 0.5 Hz can be achieved; this is valid for an antenna beamwidth of 2 degrees. The chosen C-scope type display with 1,350 pixels has the advantage of being easily interpretabel; for a technician at this state in the program.

When using a C-scope-type display, the range information normally is lost, but this was avoided by using different symbols for different range gates. Having a maximum range of 600 m, 16 symbols were chosen to distinguish range gates 37.5 m long.

The 5 x 5 -inch CRT built into the helicopter is shown in fig. 5.

FLIGHT TESTS

Different obstacles were detected during flight tests carried out in autumn and winter 1982 and 1983 in cooperation with Messerschmitt Bölkow Blohm (MBB) of Munich, West Germany.

Mechanical and electrical compatibility of the radar unit with the helicopter, MBB type BO 105, was demonstrated. Besides hightension lines, other obstacles like forrest and countryside edges, single trees, and bridges were detected.

A typical measurement situation is shown in fig. 6, displaying two high-tension lines with posts, a 220-kV line in the front, and a 5-kV line in the back. The distance to the post of the 220kV line is approximately 150 m; lines are about 30 m apart. The photograph gives an azimuthal angle of about 60 degrees.

The corresponding C-scope radar display is given in fig. 7. At the bottom the ground clutter can be seen. Due to the mechanical structure, the post reflects over the full heigth, which can be seen in the middle. To the left, four vertical lines are displayed. Due to the fact that power cables are generally of the wrapped-construction type, not only one but three line echoes occur, shown at the farther left. In addition to the specular return at normal incidence, there are returns at about ± 10 degrees to normal when the path length difference between strands equals half-a-wavelength. This principle is shown in fig. 8. The communica n wire at the top results in one reflection only, because it is a communication cable and not wrapped.

To the right of the post of the 220-kV line, the 6-kV line reflections are displayed. Displayed also are the posts of the 6-kV line, again at the left. It has to be remarked that the C-scope display shows the entire azimuthal range of 180 degrees.

The optical view of a typical flight situation is shown in fig. 9. A high tension line can be seen in front of a forrest. In the center of the C-scope display (Fig. 10) four vertical lines can be observed (line post and the line echo triple). Range information is given by the different pixel symbols.

CONCLUSION AND PROPECTS

The viability of the incoherent 60 GHz radar approach for helicopter obstacle warning was proven. High voltage transmission lines and funicular cables, i.e. wire-like obstacles, can be detected at distances of more than 400 m.

In order to perform the requirements of military helicopter missions a suitable display has to be developed. Following an audio warning, the pilot must be able to inform himself about the danger situation at one glance. The line echo triple might give the opportunity to identify a power cable by means of a pattern recognition algorithm. Thus wire-like obstacles can be distinguished from other, like posts or chimneys.

For air-speeds of more than 100 kts the information updating has to be accelerated. An updating of 5 Hz for example can be accomplished only, using electronic antenna switching for the elevation scan.

Both items are subjects under developments at the time.

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ACKNOWLEDGEMENT

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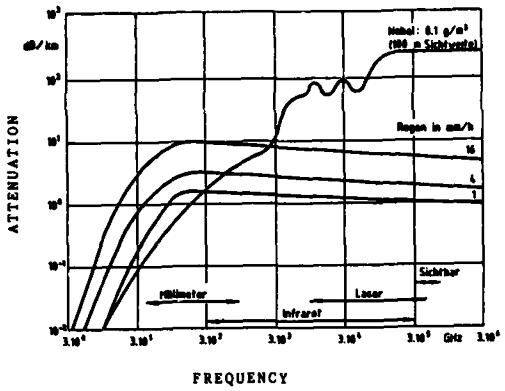

Fig. 1 Rain and fog attenuation versus frequency

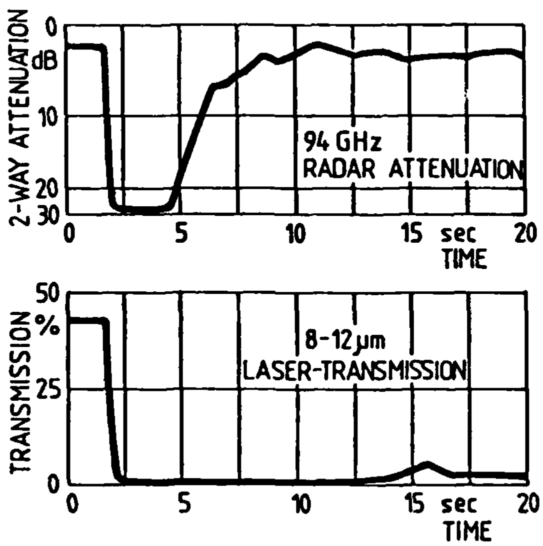

Fig. 2 Propagation for 94 GHz radar and IR-laser in a TOT-barrage

Fig. 3 60 GHz pulse radar sensor for radar cross-section measurements of power cables

Fig. 4 Scanning 60 GHz pulse radar sensor mounted to a testbed aircraft

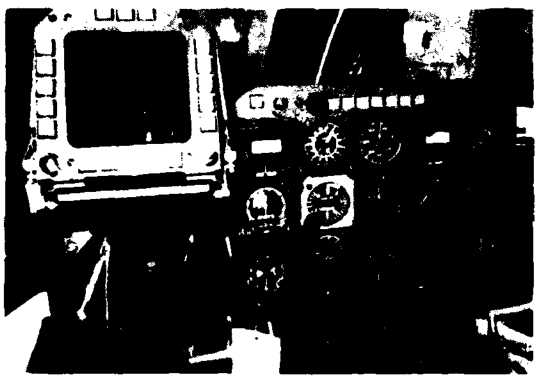

Fig. 5 Sample installation of radar display in the testbed helicopter

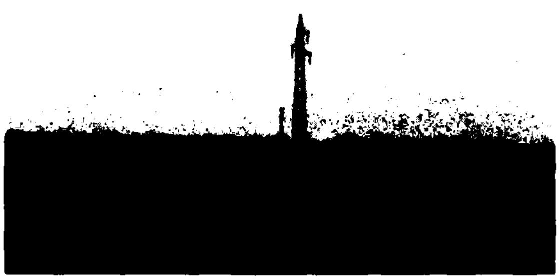

Fig. 6 The measurement situation described in the text used two sets of wires: towermounted, 220-kV lines and pole-mounted 6-kV wires. Atop the towers is a communications wire.

Fig. 7 C-scope display corresponding to the scene shown in Fig. 6 clearly shows a hazard. The display requires some interpretation to recognize that two separate sets of wires are shown, but it was designed for measurement purposes only.

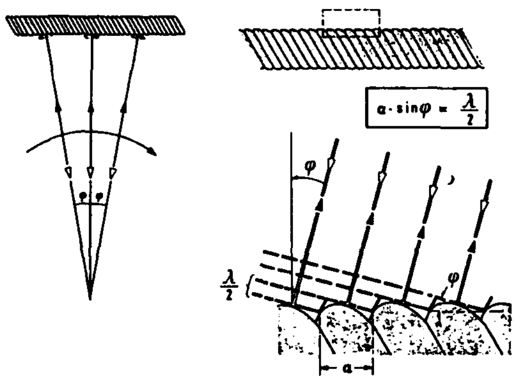

Fig. 8 Millimeter-wave wrapped wire detection principle, the line echo triple

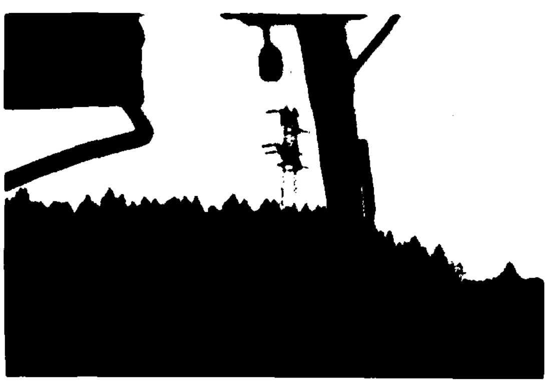

Fig. 9 High-tension line at forrest edge can be seen through the windscreen, but nearly disappear from view during inclement weather.

Angle of view is approximately 60 degrees wide.

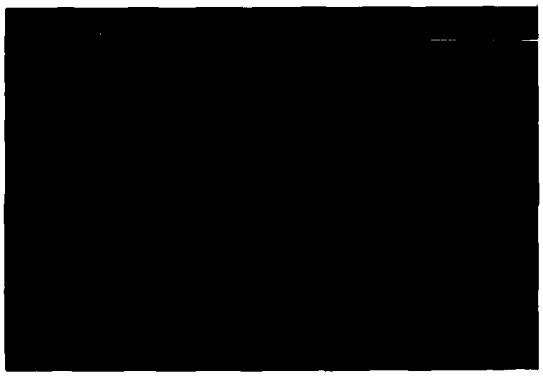

Fig. 10 C-scope display of scene shown in Figure 9 indicates range of hazard through pixel symbols; the three echoes provide serendipitous emphasis to the risk.

Display covers an azimuth of 180 degrees, providing protection in directions toward which the pilot may turn to avoid dead-ahead obstacles.

THE DESIGN AND DEVELOPMENT OF AN INTEGRATED CORE SYSTEM FOR BATTLEFIELD HELICOPTERS

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1. INTRODUCTION

Battlefield Helicopters are required to operate in particularly difficult and hostile operational environments. Crew effectiveness, aircraft survivability and maintainability are among the key requirements which need to be realised in both anti-armour, reconnaissance and tactical transport missions. To achieve these goals, a degree of integration is required to bring together the many subsystems, navigation, communications, weapons etc. in an architecture which not only enables the crew to manage and control the systems and aircraft in the battlefield environment but provides a means of extending the performance of the total system, crew, avionics and aircraft.

This paper describes the concept of the integrated systems being developed for Lynx and Westland 30 product improvements and in particular, the advanced development programme which has been underway at Westland to prove the concept of the 'core' system common to both aircraft.

2. BACKGROUND

The studies which have been undertaken to determine the problems and difficulties associated with battlefield helicopter operations have all indicated a need to reduce workload within the cockpit. Two aspects need to be addressed; firstly to reduce the number of controls and displays to reduce the pyschomotor and visual workload involved in operating controls and scanning instruments, displays, maps and associated documentation.

However, merely reducing the number of separate, dedicated instruments, displays and controls and replacing these by large electronic displays and multi-function controls will do little if indeed anything, to overcome the problem. Developments such as these will help in reducing the overall size of instrument panels, improving the external view from the helicopter, improving the readability of displays and accessibility of controls but do not have a significant effect on the crew - in fact the addition of many new avionics and weapon systems, improved navigation, countermeasures etc. can exacerbate the workload problem as far as the crew are concerned.

The second, and in our opinion most significant issue concerns the introduction of a level of automation between the crew, aircraft and systems. The system would perform many of the routine monitoring and control tasks associated with the management and operation of aircraft and systems and thus provide the crew with more time to perform the interpretive and judgemental activities associated with low level tactical flying in poor visibility conditions.

The battlefield helicopter will operate in a hostile environment, away from well equipped maintenance and support facilities and yet must be capable of providing a high level of availability. The helicopter and its avionic system must be capable of absorbing damage from hostile fire, random faults etc. and still provide some degree of operational capability.

An avionic system, designed on a piecemeal basis, will not be compatible with the type of mission which we envisage and a new architectural approach which overcomes most, if not all of these problems, is required.

In addition, Lynx and Westland 30 variants are being developed for battlefield applications in broadly similar timescales. Both aircraft have different mission requirements, the Lynx designed primarily for the anti-armour role, the Westland 30 as a tactical transport aircraft. While there are differences in mission profile, system performance, crewing philosophies, weapons and sensors etc., both aircraft share common requirements in many areas, requiring improvements in cockpits, maintainability, reliability etc. and reductions in cockpit workload.

These issues pointed to the need to develop a system architecture in a modular fashion which could satisfy the mission requirements of both aircraft. The elements of the system which are common to both aircraft is a suite of processing and control and displays units. This is referred to in this paper as the 'integrated core' system. The core system has been under development at Westland for three years. The aims of this programme have been to develop the system to a point where it can be committed to specific product applications with minimum risk, to develop the many procedural and management disciplines peculiar to software intensive systems and finally and perhaps the most compelling of all the objectives, to implement new development techniques capable of dealing with software intensive, embedded computer systems.

3. THE INTEGRATED CORE SYSTEM

The concept of the core system is shown in Fig. 1. The design aim is to integrate the subsystems required for anti-armour and utility battlefield operations around a common 'core' of processing and control and display facilities. Generally the anti-armour and utility missions will require identical systems apart from the target acquisition and weapon sub-systems which are peculiar to the anti-armour role.

The core elements of the system are required to perform a number of functions dealing with both mission and aircraft aspects, the main objective in both areas is to reduce workload associated with management and control of aircraft and mission sensors.

The primary mission mangement functions which are performed are communications and navigation management, control of defensive aids and identification aids. The 'basic' aircraft functions include flight, performance and fuel management and comprehensive maintenance 'surveillance' including avionics test and diagnostics, aircraft health, usage and condition monitoring. These facilities are briefly described in the following paragraphs.

3.1 Communications Management

The communications management function provides mode control, frequency selection and status monitoring of all radios and IFF. The system is designed to contain up to fifty preset frequency sets, each set consisting of the frequency, nickname and frequency dependent mode selection data.

3.2 Navigation and Management

The navigation functions utilise information generated by both Radio Navigation aids such as VOR/DME and ADF, and autonomous facilities including Doppler, compass and air data to provide a continuous best estimate of geographic position. Provision for other sensors has been made as a number of mission scenarios require significant improvements in navigation accuracy.

In the battlefield there is considerable scope for improving crew effectiveness by including extensive flight planning functions within the system. A flight plan would include, typically, data on target and weapon locations, ground obstacles, hazards, avoidance areas and cover routes.

Flight plans can be generated by either calling up a pre-stored route by name, or by calling up a number of pre-stored waypoints or a combination of both together with impromptu, manually entered waypoints.

An automatic flight direction function provides steering or guidance information, to the pilot enabling the aircraft to be flown along the Flight Path.

An optimisation function is also provided, this, operating in conjunction with the navigation and planning function, enables the crew to establish the distance between two waypoints, overall route length, time of flight and estimated time of arrival at selected waypoints or locations.

3.3 Performance Management

The performance management facilities provide a range of performance calculations for both normal two engine conditions or one engine inoperative conditions. These include;

- Cruise calculations
- Hover calculations
- Power available
- Climb performance
- Autorotation envelope
- Time on station.

The performance and power monitoring facilities permit the pilot to exploit the maximum available performance from the aircraft, overcoming the limitations imposed by conventional placard or generalised restrictions.

3.4 Maintenance Surveillance

A comprehensive set of maintenance facilities have been specified. These are discussed in some detail in ref. 1. and include:

a) Engine power monitoring
b) Engine usage monitoring
c) Transmission exceedance monitoring
d) Transmission usage monitoring
e) Rotor usage monitoring.

The objective of these functions is to relieve the pilot of parameter exceedance monitoring tasks and, by processing the range of data within the system, provide more readily understood warnings of impending system failures and limit exceedances. Provision of component defect assessment and life consumed information will contribute to an increased usable life in major aircraft components, an increase in Mean Time Between Removals, reduced maintenance workload and reduced aircraft down time.

At the avionic system level the core system performs a number of Built in Test (BIT) functions. Processor and memory testing is performed periodically to confirm the functioning of the of the processors, direct access memory and local logic.

Each processor unit also tests the various sub-system interfaces and together with the BIT status determines the overall operational condition of each LRU in the system. This information is used to initiate the system reversionary modes and also to generate LRU test/status page data to the crew to be viewed when convenient.

3.5 System Architecture

The core system architecture is based on a dual processor system connected to the cockpit control and displays via a MIL-STD-1553B data bus. These processors, in addition to providing the list of aircraft and mission functions listed above, provide bus control facilities. Two additional equipments are provided, an Emergency Control Panel (ECP) which provides for the direct (emergency) selection of radio frequencies and IFF codes and erasure of confidential information stored in the system. A Data Transfer Device which provides the means of entering pre-flight data such as flight plans, codes, nicknames etc.

Systems and equipments are interfaced to the core system in one of two ways, either directly to the data bus, or for those equipments and system which do not have a bus interface, connected directly into one of the two processor units [see Fig. 2].

3.6 Redundancy and Reversionary Capability

Sufficient redundancy and reversionary modes of operation are provided in the mission to ensure that no single malfunction will result in a mission abort. Reversionary modes include:

- Overall system control from either CDU.
- 'Mission critical' functions are duplicated in both processor units.
- In the event of a total systems failure, emergency radio and limited navigation and IFF facilities can be manually selected.

3.7 Spare Capacity

To ensure that the system is capable for future expansion to meet either a customer's evolving mission requirement or new aircraft developments, considerable spare capacity has been designed into the system.

This spare capacity is provided for in three ways.

Accommodating additional functions within the two processors will require additional processing capability. How much processing will be required is difficult to predict, but the rule of thumb used in the existing design was to allow for a 50% spare capacity in processing time available. We also anticipate a significant growth in memory requirements during the development of the system and subsequent service life. Growth factors of two to three over the basic memory allocation are not considered unreasonable and the system is designed to accommodate this.

The data bus utilisation is currently 10% and so there is a significant spare capacity to accept additional equipments and the bus traffic which they will generate.

Many equipments are not compatible with MIL-STD-1553B either because they were designed prior to the standard being introduced or because it is uneconomic to introduce a fairly complex and relatively expensive interface into a simple equipment or sub-system, there will be a continued requirement for equipment to be interfaced directly to the core system processors. Spare card positions are provided to accommodate growth in this area.

4. RIG DEVELOPMENT PROGRAMME

The integrated nature of the system, together with the software intensive aspects, necessitated the adoption of new techniques and facilities for verification and validation. Software studies have shown that the earlier that system errors are detected the lower the cost of correcting the error. The importance of a thorough and effective test phase is emphasised by comparing the relative costs of correcting the error. The importance of a thorough and effective test phase is emphasised by comparing the relative costs of correcting software errors during the early prototype phase with the costs of in-service modifications. The latter can cost up to twenty times more than the early development correction. [Ref. 2].

Over a period of some years, considerable investment had been made in improving simulation facilities within the company and studies which were undertaken suggested that these simulation facilities had been developed to a stage such that they could be used in lieu of actual flight testing in some areas. This is important for two reasons. The integrated system has real potential for creating complex test problems which could be difficult to identify and solve in an airborne test environment. Ground testing, however, with well equipped facilities can provide substantial benefits and improvements in productivity. The ground test environment provides the capability to obtain precise repeatability, instantly halting or 'freezing' the system in a particular mode of operation and re-running tests. Finally the economic argument for improving the quality of ground testing is a powerful one. Ground testing can be achieved at between a one fiftieth and one hundredth of the cost of flight testing and in a reduced timescale.

The integrated core system demonstrator programme was initiated with the following aims.

- Test, verification and assessment using ground based test techniques,
- Analysis and support of flight trials,
- Development of production test techniques and facilities.

The general philosophy has been to devise a test approach which proves the system through a series of tests, initially testing each function independently to confirm the operation, then progressively adding groups of functions to examine the more complex aspects concerning the interaction of functions and the behaviour of the system under fault conditions.

A test rig was required to support the development programme, the principle of the rig being to simulate the core system by injecting signals and monitoring the response of the system to these stimuli.

The main criteria which determine the size, complexity and cost of the test rig are:

a) Test run time; length, real time factor,

b) Simulation quality and quantity, whether the simulations provided will be authentic or 'simple',

c) Period within which all of the selected simulators should be iterated,

d) Amount and content of stored data,

e) Extent of analysis required.

The overall block diagram of the rig showing its relationship to the core system is shown in Fig. 3. The test rig is based on a centralised processor, providing control, monitoring and various analytical tools necessary to support the development programme. Included within its facilities are multi-user operator terminals, hard copy, disc and tape storage etc.

A substantial part of the task of the rig is the support of simulation and emulation functions to enable testing to take place in the absence of one or more of the sub-systems. This facility enables the core system hardware and software to be exercised in a variety of simulation mission situations with a high degree of realism.

Model and analysis functions have been implemented by structured top down design in Pascal and contained within 300K bytes of memory. The design contains less than 100 modules each consisting of no more than 150 lines. The entire program has been run on a desk top mini populated with 1M byte Ram and supported by 16M bytes hard disk store. This configuration provides the facility for a continuous simulation run of at least 3 hours with full data logging and analysis.

The test programme has addressed the following:

a) Verification of core system and sub-system interfaces. Detailed testing to validate MIL-STD-1553B bus interfaces and the equipments which have been hardwired directly into the system.

b) Functional aspects (computational effectiveness and accuracy).

An assessment of the functional aspects of the system and the achieved computational accuracy of the core system is achieved by exercising a 'closed loop' dynamic test. The performance of the system is monitored by means of analysis software contained within the rig, this information can be stored and compared with data obtained from later tests.

c) EMC and Power System Compatibility

By ensuring that the equipment is installed on the rig in a similar manner to the aircraft, a considerable amount of EMC testing can be carried out off aircraft. Bonding, earthing, wiring and RF emissions can be assessed. It is also necessary to connect the rig to representative aircraft supplies for test runs to assess loading, switching and transient effects.

d) Reversion and Failure Mode Investigation

The rig is used to examine the complete range of failure modes and reversionary modes designed into the system by generating the appropriate fault patterns and analysing the system response. This facilityis also used to verify the performance of the built-in-test system and diagnostics.

e) Human Factors Assessment

The aim of the rig is to provide facilities to enable the complete man-machine interface to be assessed. The full potential of the system will only be realised if the man-machine interface is correctly configured. The test programme addresses both the physical characteristics of the controls and displays - size, shape, location, brightness etc. and the capability of the system to undertake the routine monitoring and control tasks.

The full impact of this type of facility has yet to be assessed but we believe this type of dynamic test rig will be essential to the timely and economic deployment of future digital avionics. The use of simulation as part of the development process can verify and lend more confidence, accuracy and comprehensiveness. As an example the following table indicates the way in which the rig performs key functions.

TASK	Percentage of Testing	
	Rig	Aircraft
Software Verification	98	2
EMC	85	15
Crew Familiarisation	95	2
System Integration	80	20
Failure Modes and Effects	90	10
Power Systems Compatibility	85	15

The general arrangement of the laboratory showing test rig equipment bench, general purpose instrumentation and computing is shown in Figure 4.

5. CONCLUSIONS

The integrated core system is one of the key technologies which will improve the effectiveness of anti-armour and tactical transport helicopters by removing the need for the crew to perform many 'in cockpit' monitoring and control tasks.

The system is designed to incorporate both mission and 'basic aircraft' functions, providing facilities to enable the crew to exploit the full performance of the aircraft and systems during a mission and facilities to improve maintainability of both aircraft and systems.

The integrated nature of the system has necessitated the introduction of new tools to test and validate the behaviour and correct functioning of all the elements of the system. A highly software intensive system also requires strict management disciplines, quality standards design and development disciplines be implemented at the outset of the programme.

The advanced development programme undertaken at Westland has successfully proven the practicality of the systems - that a highly integrated system can be cost effective and can meet differing role requirements and can be developed within an acceptable timescale.

The development of high quality, high performance development tools requires a significant effort but promises to provide significant savings in aircraft and system development programmes.

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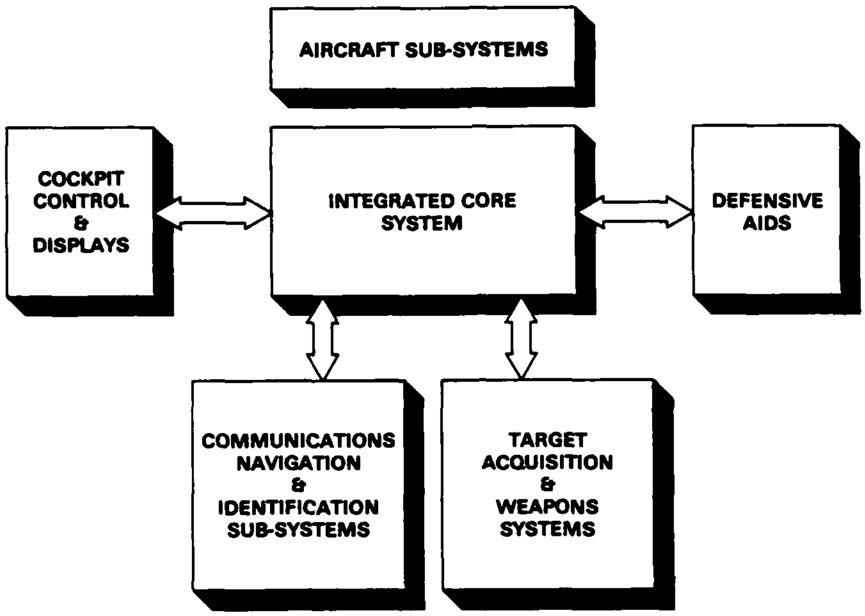

FIGURE 1 THE INTEGRATED CORE SYSTEM CONCEPT

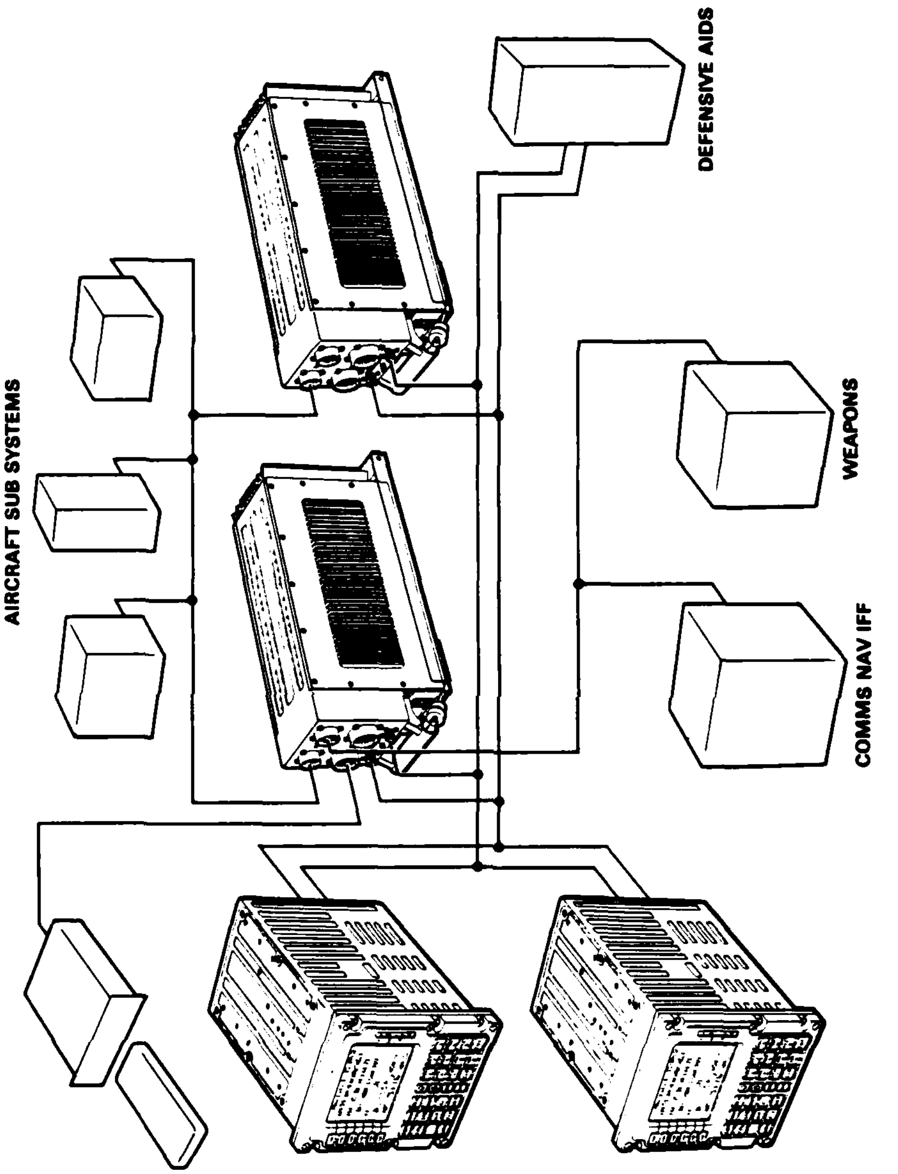

FIGURE 2 INTEGRATED CORE SYSTEM

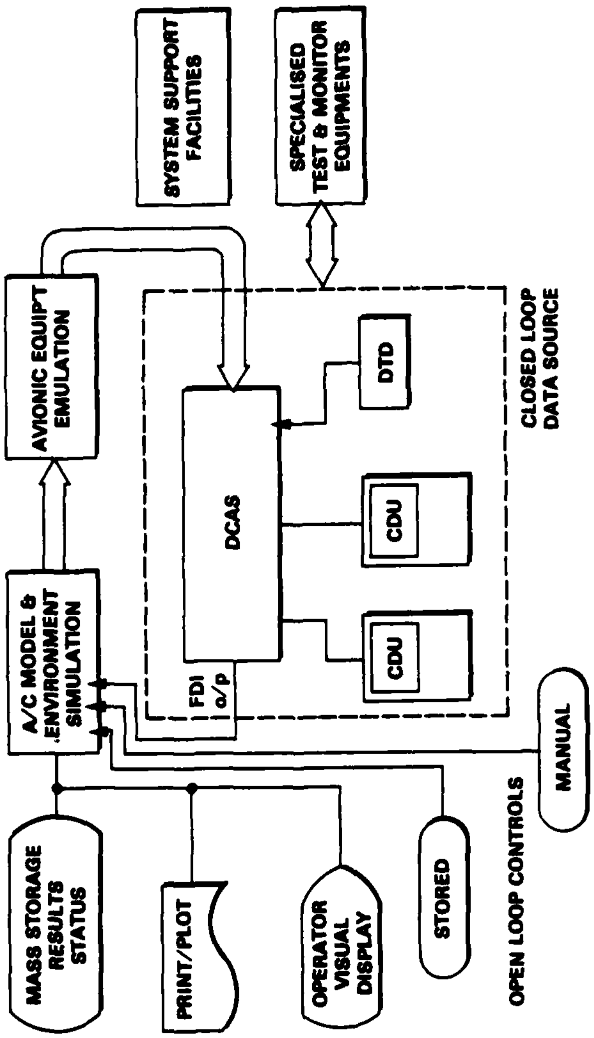

FIGURE 3 CORE SYSTEM TEST RIG ARCHITECTURE

FIGURE 4 GENERAL ARRANGEMENT OF CORE SYSTEM INTEGRATION RIG

EVOLUTION TOWARD A MULTI-BUS ARCHITECTURE FOR ARMY HELICOPTER AVIONIC SYSTEMS

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SUMMARY

Over the past decade the architecture of Army helicopter avionic systems has evolved from stand-alone subsystems to integrated systems characterized by digital data buses and embedded microprocessors. As these first generation integrated systems are reaching maturity, the U.S. Army Avionics Laboratory is turning its attention to some of the issues which must be addressed in preparation for the development of future systems. This paper presents an evolving concept for a data bus and processing structure which is being developed under an in-house Avionics Laboratory technology base effort. This concept will be examined extensively under the next several years using contractor and in-house developed hardware and software tools.

As a result of the need for significantly increased integration of aircrew functions to meet the demands being placed on Army Aviation, it has become apparent that a technical requirement exists for an architectural concept with a number of characteristics that go beyond that which is in development today. For example, many of the subsystems that come together to synthesize a total helicopter avionic system are developed by organizations with specific expertise in various functional areas (e.g., navigation, flight control, weapons, target acquisition. etc.). These organizations are usually both geographically and managerially separated, both in the government (customer) and in private industry (supplier). It has become apparent that an architectural concept is required that will allow these various functionsl areas to develop both subsystem hardware and software independently and still conform to an architecture that facilitates future total system integration. Similarly, subsystem requirements in the control/display area must be developed with the subsystem. However, the architectural concept must allow subsystem control/display functions to be mapped into the total helicopter control/display system without re-doing the associated software. From the subsystem developer viewpoint, the system architecture must be such that at the time of total system integration, subsystem specific hardware (e.g., sensors) must be easily incorporated into the bus structure and the system level software absorbable into the system processing elements.

Another desirable feature or characteristic of the new architecture would be a means for high speed interface between processing elements of the system. Additionally, both the processing structure and bus structure must lend itself to fault tolerant designs.

Finally, there are two remaining areas which the architectural concept must address if it is to be widely accepted. First, the hardware implementation of the concept must be such that it lends itself to a procurement process that does not unnecessarily restrict the government and contractors to either each other or specific technology; and second, the processing elements must be such that standardization can be achieved on a foundation of structured programming and a higher order language.

Considering all of the above characteristics, an architectural concept is evolving based on multi-busses and multi-processors. This paper presents this architectural concept.

1. INTRODUCTION

At a paper presented at the 32nd Symposium of this Panel (Referenced), the step-by-step approach toward system integration taken by the U.S. Army Avionics Laboratory was presented. To that point, emphasis was on achieving integration via a multiplex data bus and multifunction controls/displays. As these first generation integration concepts are being implemented in helicopters such as the Army OH-58D and the Air Force HH-60D, attention is now being focused on evolving toward hardware and software architectures that will allow for total system integration to take place in less time and with reduced costs. However, a great deal of the constraints present today will remain into the foreseeable future. Therefore, the architecture evolved must be able to achieve the high levels of integration required while at the same time being subject to these constraints. The purpose of this paper is to present the evolution that is occurring in the area of system architectures and then evolve further toward a concept that will accommodate the significantly increased level of integration required if helicopter aviation is to achieve the demands of postulated future missions. In addition to satisfying the in-place constraints, therefore, the evolved concept must be flexible in that the architecture can accommodate future technology such as Very High Speed Integrated Circuits (VHSIC) hardware, and sensor blending and expert system software.

2. EVOLUTION TO DATE

It is most helpful prior to presenting an architecture to first look back in order to trace the path and review the thought processes that led to our current architecture. So like White Rabbit in Lewis Carroll's Alice, let's "Begin at the beginning."

An accepted beginning to this era of integrated system architectures is the promulgation and acceptance of the U.S. MIL-STD-1553 data bus (NATO STANAG 3838). For ease of discussion, let the line depicted in Figure 1 represent such a dual (redundant) twisted shielded pair data bus. The subset of equipments which the Army Avionics Laboratory first used the bus concept to integrate was the communication, navigation, and identification (CNI) subset (see Paragraph 2 of Reference, "Integration of CNI: The First Step"). Architecturally speaking, the integration of this subset can be represented by Figure 2. Generally speaking, all processing was accomplished in 8-bit microprocessors embedded in Remote Terminals (RT's) which also

served as interface media to the mostly existing inventory radio equipments. One or more control/display units (CDU's) provided means for crew interface. Architecturally, the system can be characterized as of low bus data rate, modest level of fault tolerance (e.g., completely redundant processing), and little synergism. The software was in assembly code and generally not portable. Spinoffs of this system developed by the Army are currently being used in a large variety of aircraft (such as C-130, SRR, MRS, A-10, KC-135, F-111, EA6B, etc.).

The next step in system evolution was an expansion to include helicopter functions such as flight displays, engine displays, caution/warning/advisory subsystems, electrical systems (circuit breakers), and a large number of controls/displays referred to as secondary systems. This step was accomplished by addition of the items shown in Figure 3. The crew interfaces for the functions absorbed are four multi-function displays (MFD's) and two keyboard terminal units (KTU's). Eight remote terminals provide interface to the many hundreds of aircraft sensors, transducers, etc. Generation of alphanumerics and vector graphics are accomplished in two programmable symbol generators (PSG's). The processing elements consist of two Sperry SDP-175 16-bit processors. Programming is still generally in assembly code; however, concepts such as structured software with software modules are evolving. Architecturally, the system can be described as of modest data rates, fully redundant, and through use of dynamic bus allocation (DBA) somewhat partitioned software (i.e., for a fixed time period of every frame the master bus controller in one of the SDP-175's relinquishes the bus to the CNI subset at which time all CNI traffic is accomplished). This architecture (with some hardware optimization which eliminates the need for DBA) is currently being applied to systems such as the OH-58D and the HH-60D. The Army Avionics Laboratory in-house efforts (see Paragraph 3, "The Army Digital Avionics System (ADAS): of referenced paper) is now adding Voice Interactive Avionics (VIA), both synthesis and recognition, and a digital data link, the Airborne Target Handoff System (ATHS), to the system (see Figure 4).

3. THE NEAR FUTURE

In addition to adding these new functions to ADAS, it has become apparent that through additional processing a much higher level of automation can be introduced into this system. If it is assumed that this new processing is to be implemented using the new DoD standard Higher Order Language (HOL), Ada, and the processor used is to be characterized by a standard Instruction Set Architecture (ISA), such as MIL-STD-1750A for 16-bit machines or MIL-STD-1862 for 32-bit machines, then the addition of a processor as shown in Figure 5 becomes very appealing. (For illustrative purposes only one processor is shown; however, in reality there would be redundancy.) Now further assuming that all the necessary software tools are mature enough to distribute to the engineering elements that have cognizance over the subsets of this system, the processing functions currently embedded in the CNI subset and the SDP-175's can be placed in software modules which can then reside in the new processor. Only a modest amount of processing would remain outside this new processor (input/output, signal conditioning, symbol generation, etc.).

If attention is now turned to the architecture evolving in the navigation technology area, a virtual step-by-step analogy will occur resulting in expansion of the system to that shown in Figure 6. Similar to the basic aircraft area, by the addition of processing, a much higher level of synergism can be achieved among the navigation elements. The processor shown at the bottom of the navigation sub-set data bus in Figure 7 is, of course, identical to the basic aircraft processor just discussed (HOL, ISA, structured programming, etc.). In actual fact, they could, of course, be the same processor or maybe two processors residing in the same box interconnected by a high speed bus (see Figure 8). Nevertheless, the important point at this time is to note that all system level processing software will be in a higher order language (Ada) and will be in modular form.

A further analogy (Figure 9) can now be made for the system level processing in the various functional areas such as flight control, fire control, stores management, target acquisition, visionics, aircraft survivability equipment, etc. That is to say, in each of these areas, system level processing functions would be developed using the same software tools and the same software structure.

Now each of these functional areas will require unique crew interface actions which, although developed separately, must be mapped into the total cockpit. During development of these various subsystems, some crew interface software and hardware must be developed and used to accomplish many of the necessary steps in subsystem development. It is important that in this subsystem development process the control/display hardware used and the software developed for it be capable of being mapped into a totally integrated cockpit (i.e., minimize hardware uniqueness/use software modules). Now assuming a separate cockpit data bus (see Figure 10), and for illustrative purposes another processor, the crew interface software from the various subsets can now be mapped into this processor at the time of total system integration.

A structure has now been created which allows the various functional areas to develop hardware and software independently, albeit under certain constraints, yet also prepares for total system integration. Briefly stated, the constraints would be: to write all system level software in an HOL in accordance with a priori established rules, eliminate system level processing in the various sensors, and to use generic control/display hardware during development that is a functional subset of this total cockpit. Further, bus concepts that are evolving would treat the data bus interface as any other interface tied to the multiprocessor global bus thereby making the data bus transparent to future high speed data bus standards. As shown in Figure 11, the various data buses would be geographically distributed on an airframe at the time of total system integration even though development of the subsystems used functional distributions. Finally the multiprocessor would exhibit certain characteristics such as a global memory tied to the global bus, local memory with each microprocessor, and a very high speed interface by which a global bus in one multiprocessor can be tied to the global bus of another multiprocessor which is executing other functions or possibly the same functions because of fault tolerant considerations.

4. A FEW YEARS HENCE

The multiprocessor/multibus architecture evolved is in essence technology transparent; however, a few points must be made with regard to further evolution that includes hardware technology (such as VHSIC chips) and software technology (such as sensor fusion algorithms). Assuming that some of these processing

functions will be located in special modules (e.g., signal processors) also tied to the processor global bus, it becomes necessary to create a means whereby the very wide bandwidth data from the sensors can be fed to these modules. This can be achieved via direct ports to the multiprocessor or if fault tolerant designs are to be achieved a very high speed sensor data bus may have to be synthesized. Much thought is currently being focused on this area and a number of concepts are evolving. The U.S. Army Avionics Laboratory is pursuing a number of studies to determine which of the concepts are applicable to helicopter systems.

5. CONCLUSION

The concepts presented in this paper represent an attempt to arrive at an architecture which fits the avionics technology available today and that which will be available in the near future. Current efforts to implement this technology in the U.S. Army Avionics Laboratory are using currently available microprocessors configured as a multiprocessor and an available Ada compiler. A procurement strategy has evolved in parallel with this architecture that uses form, fit, function specifications and interface control documents so as not to restrict future procurements to today's technology. Certainly much work remains to be accomplished to evolve the concept presented here; however, it is fully expected that over the next several years, enough experience will be gained to achieve an architecture that will meet the needs of the demanding helicopter missions of the future.

REFERENCE

Dasaro, J. A., Elliott, C. T., "Integration of Controls and Displays in U.S. Army Helicopter Cockpits," Proceedings of 32nd NATO Guidance and Control Panel Symposium on The Impact of New Guidance and Control Systems on Military Aircraft Cockpit Design, Stuttgart, Bad-Cannstatt, Germany, May 1981.

Figure 1. "The Beginning" - STANAG 3838 Data Bus (MIL-STD-1553)

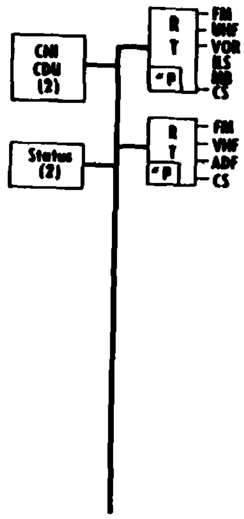

Figure 2. CNI Subset: The First Step

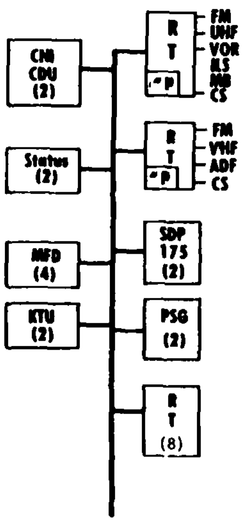

Figure 3. Expansion to Include Flight/Engine/Caution/Warning, etc.

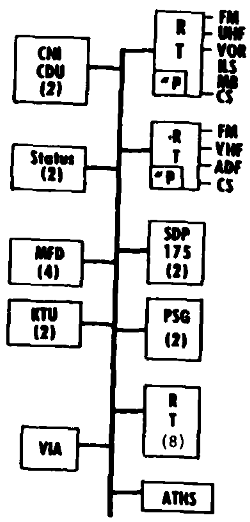

Figure 4. Voice Interactive Avionics and Airborne Target Handoff System Added

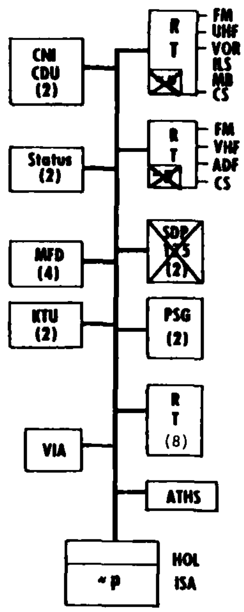

Figure 5. A Processor Characterized by Standard HOL and Standard ISA

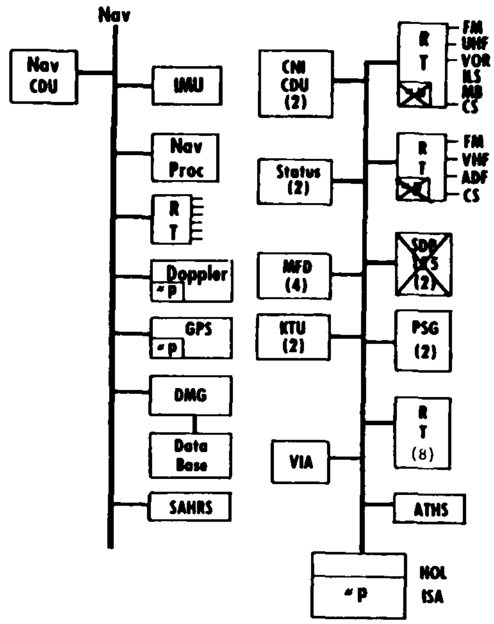

Figure 6. Navigation Technology Architecture

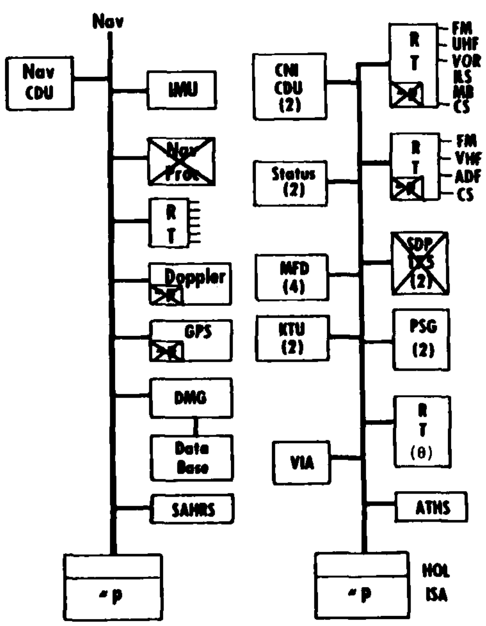

Figure 7. Navigation System Processing

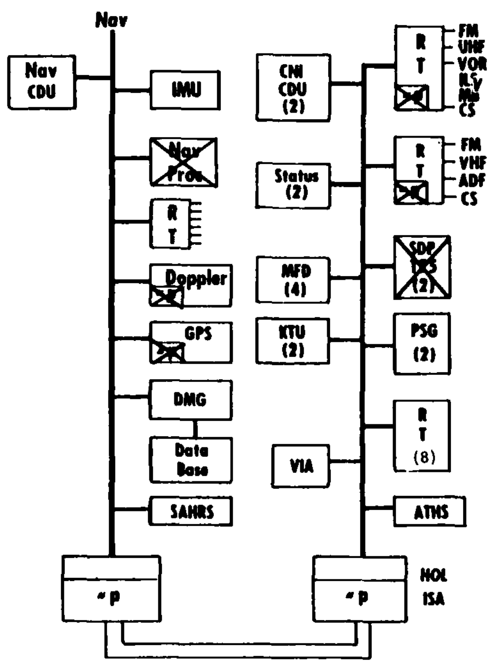

Figure 8. High Speed Bus

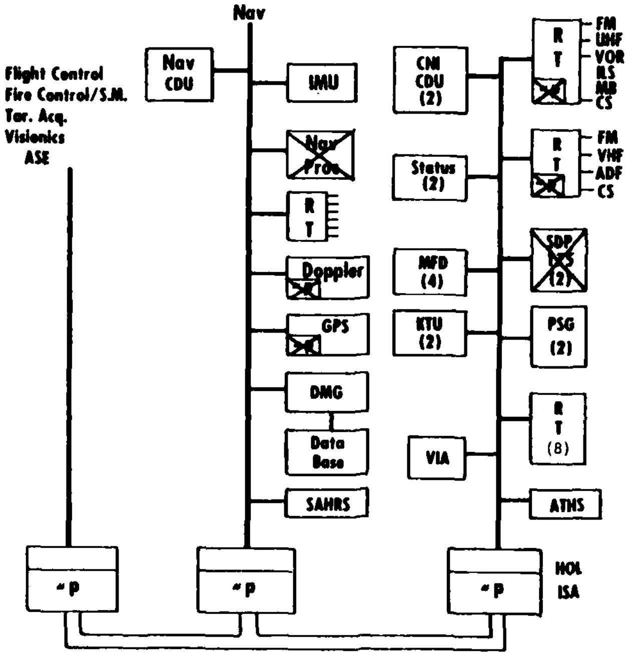

Figure 9. A Further Analogy

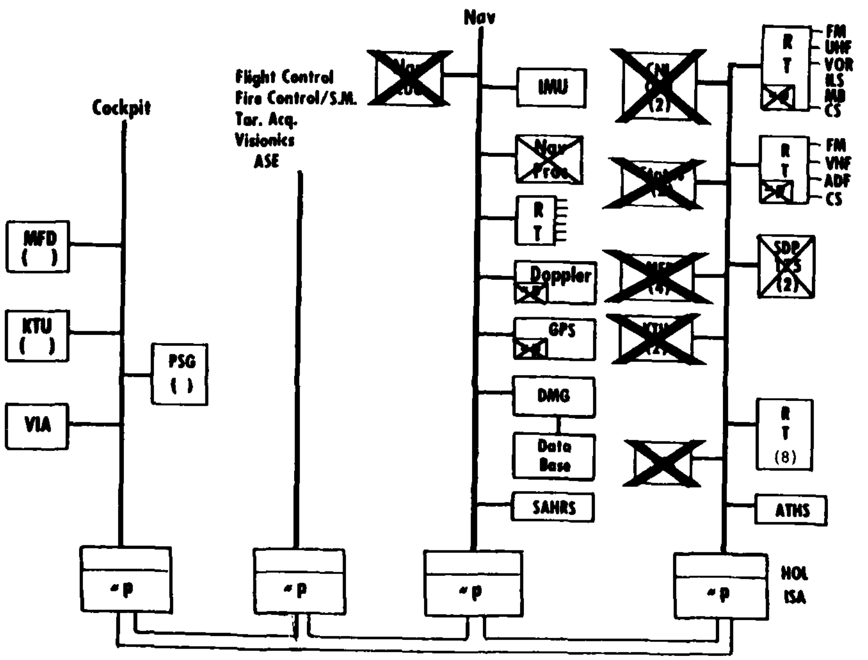

Figure 10. Mapping Crew Interface Functions from Subsystems to Total System

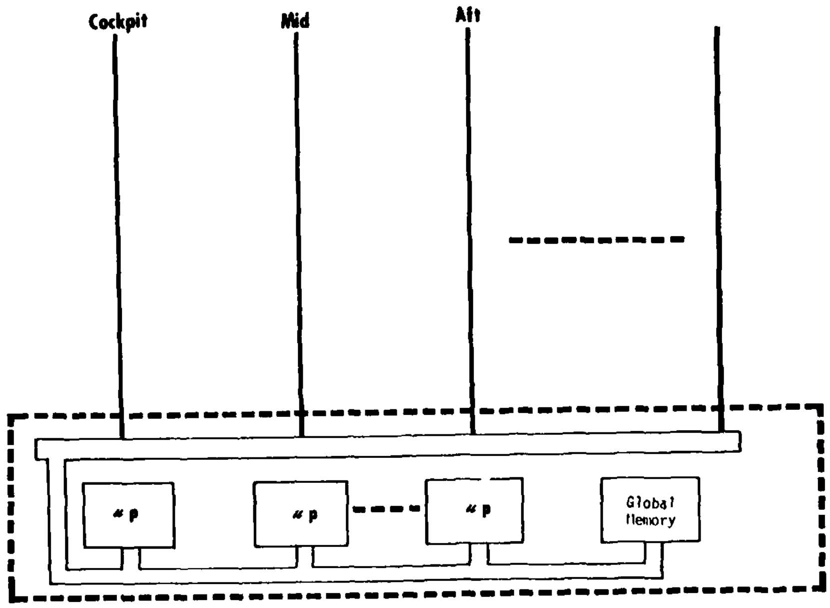

Figure 11. Multi-Processor/Multi-Bus Architecture

INTEGRATION OF SENSOR FUSION IN ADVANCED HELICOPTER COCKPIT DESIGN

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SUMMARY

Effective operation and survival in the battlefield of the future will impose great demands on helicopter weapon systems and the crews who manage them. Among these will be: time to detect and respond to threats and targets; ability to integrate many types of sensor data and assess the intelligence portrayed by these sensors; requirements for highly trained systems management personnel who can process highly technical information while operating in the very demanding NOE environment.

This paper will discuss helicopter cockpit designs for the future and how crew performance can be improved. Subsystems will be addressed which will improve human performance. Discussed in particular will be sensor fusion systems, related misconceptions, and their proposed use.

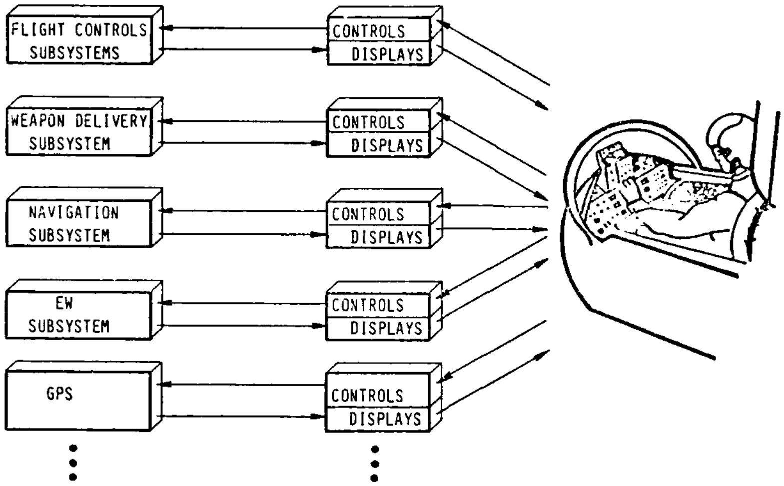
1. INTRODUCTION

If helicopter weapon systems are to remain a viable part of the army force structure on the modern battlefield, their survivability and operational capability must be significantly improved. Survivability will continue to depend on effective nap-of-the-earth (NOE) operations, signature reduction, and improvements in detecting and countering threat systems. Improvements in operational capability can come from weapons improvements such as longer ranges, shorter time lines, day-night-all weather capability, higher kill probability, and corresponding improvements in target acquisition systems. All of the equipment needed to provide these improvements must, of course, be simple and reliable.

In the past, as new requirements evolved, new systems were added to satisfy the requirements. The typical new system would have a sensing element, a processing or computing element, and a display or actuating element. For example, as new RF threats were identified, new antennas, receivers/processors, and displays were developed for the detection, classification, and presentation of the threats. When the additional workload of these new systems overloaded the ability of the human operator, a new operator was added. He was called an Observer, Defensive Systems Operator, Gunner, etc. Additional crewmembers result in larger aircraft which cost more, are easier to detect, and, as a result, require more sophisticated defensive systems and longer range offensive systems which are heavier and cost more, resulting in larger aircraft...and the snowball continues to grow. It is fairly obvious that innovative system design approaches must be applied to new helicopter weapon system designs in order to reverse this trend - to meet the requirement to operate effectively in the day/night/all weather/NOE high threat environment while reducing the size of the aircraft and the demands made on the crew. Several emerging electronic technologies when appropriately combined show promise of meeting these highly conflicting requirements. Among these technologies are further microminaturization of electronic hardware, higher speed integrated circuits (VHSIC), improvements in RF and electro-optical sensing systems, and artificial intelligence processing techniques. One of the most attractive approaches to reducing our problem is a concept called "sensor fusion."

To help understand the concept of sensor fusion, it is appropriate to start with the "classical" helicopter weapon system block diagram as depicted in Figure 1. In the figure, it will be noted that each functional subsystem such as navigation, flight control, etc., has its own sensors, processors, displays and actuators. Such a system is

FIGURE 1
CURRENT AVIONIC INTEGRATION METHODOLOGY



heavy, expensive, unreliable, and very demanding of crewmember attention. In the sudden world of the NOE flight regime where one or two seconds of inattention can result in a crash, the crew cannot afford to devote his/their time to scanning and extracting information from a large number of complex displays. The time required for human reaction may also be too long to respond effectively to threats which will be encountered on the modern battlefield. Sensor fusion offers the opportunity to reduce dramatically the number of displays required to present the information obtained from a number of sensors, greatly reduces reaction time, and makes it much easier to present to the crew the vital information extracted from the vast quantity of irrelevant or unimportant data available in a modern multisensor system. Figure 2 shows a sensor fusion implementation of the same functional block diagram which was shown in Figure 1. One major difference is immediately apparent when comparison is made to Figure 1. In the sensor fusion system, the number of displays has been dramatically reduced. A second, but less obvious, major difference is the reduction in the amount of data being presented to the crew as a result of the preprocessing to separate the information from the data and the closing of an automatic, artificial intelligence, loop around the crew to permit the automation of many of the functions which formerly required crew intervention. The sensor fusion approach will present less data for the crew to sort out, more information which the crew can act upon, and the system will carry out many of the actions quickly and automatically, advising the crew as to the action being taken. If the crewmember chooses to intervene he may do so.

The remainder of this paper will be devoted to a more detailed discussion of the sensor fusion concept and its ramifications in the modern helicopter weapon system cockpit.

2. COCKPIT DATA REQUIREMENTS

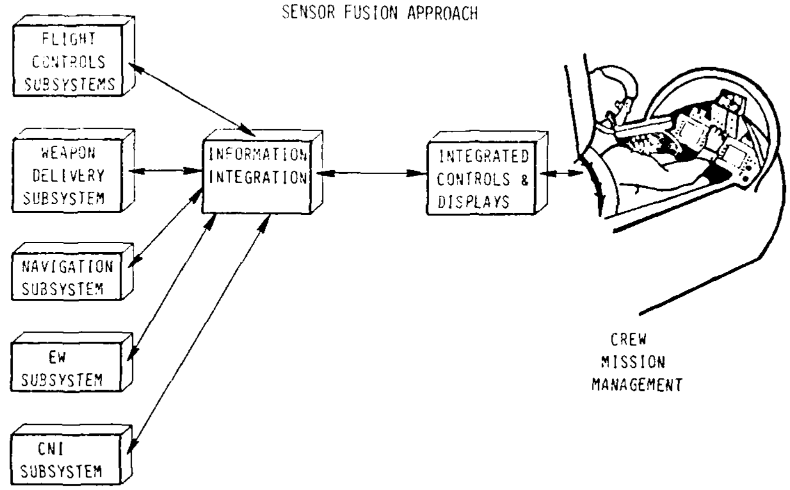
Cockpit information requirements continue to grow in quantity as the missions become more complex and as technology makes more types of information available. An example of this is the number of controls seen in helicopters has increased from 5 to 420 in 40 years (see Figure 3).

Information requirements also continue to rise as the missions demand increasingly faster response times from the human and the helicopter due to the sophistication of threats and weapons response time. In essense, as the need for cockpit information increases and the response times to those data decrease, the time available for the crew to process the information, make decisions, and implement the decisions continues to lessen. Usually it is the human's response time that is squeezed by these demands. Since human capabilities do not change, it is the human in the cockpit that requires help. Without the right kind of help, we are approaching his saturation point. Much of the obvious help can come from automation...from robotics and artificial intelligence in cockpit design.

Limited automation exists in today's cockpits. It is manifested in the form of stabilized controls, fully automated controls, and smart displays. More specifically, these include autopilots and stabilized control systems; automatic rpm controls; prioritized messages such as the caution-warning displays and flight directors and instrument landing system displays.

Cockpits of the future will certainly employ much more automation. Designs for these cockpits will not resemble the cockpits of today. Future cockpits will more

FIGURE 2
SENSOR FUSION APPROACH



nearly resembly computer terminals. The pilot will not be the tracking controller-operator he is today, using all his extremities in tracking tasks (feet on the rudder bars, hands on the cyclic, collective and throttle controls and at times even his head is used to track targets). In future cockpits, the pilot's role will more nearly resemble that of a manager rather than that of a vehicle controller.

3. PILOT WORKLOAD

The pilot's workload is often perceived to exceed his capabilities. When this happens his normal recourse is to reassess his priorities and to eliminate some of the less important tasks. Exceedingly high workload of the helicopter pilot flying nap-of-the-earch was confirmed in a study sponsored by the U. S. Army, AVRADA, Avionics Labs at Fort Monmouth, New Jersey in 1980, known as ADAS (Army Digital Avionics System). This study surveyed 34 experienced NOE Army Aviators. The results of the survey indicated the following workload situations during NOE flight:

1. The pilot uses 98% of his visual time with tasks outside of the cockpit. His manual control channels are 100% loaded with controlling the helicopter.

2. The copilot spends 70% of his time navigating.

3. Monitoring of engine and aircraft system trends is reduced to only 3% of the copilot's visual time.

These are but a few of the findings but provide an example of the high workload situation experienced by helicopter flight crews.

4. NEW HELICOPTER COCKPITS REDUCE WORKLOAD

Certain new helicopter cockpits reflect the advances in avionics which have made the integrated cockpit feasible. These advances include: the processing and displaying of a large amount of data, the use of flight computers and multiplex systems, incorporation of multifunction displays and control-display units.

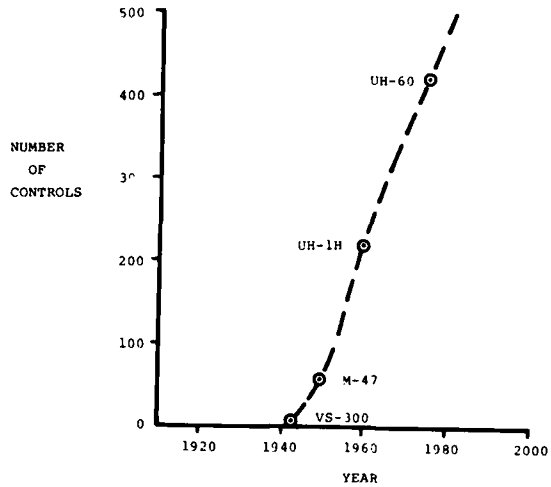
As an example, the Bell Model OH-58D (of the Army Helicopter Integration Program - AHIP) has employed these advances in avionics and with careful human factors detail design in display and control configuration, the flight crew workload has been reduced to an acceptable level for the scout mission. As an example, the typical task time for a specified navigation task was 16 seconds in the conventional OH-58, equipped with a doppler navigation system, whereas the OH-58D requires only 6 seconds after portions of this navigation task had been automated.

Cockpits of the future must employ even further sophistication for reducing workload if they are to encompass the continuing growth indicated in Figure 3. It is even more important if we are to achieve the goal of providing all of this information for the single-pilot cockpit.

5. REDUCING WORKLOAD WITH SENSOR FUSION DISPLAYS

One of the techniques for unburdening the pilot is to provide him with a capability which combines various pieces of discrete information into an integrated display for the pilot. This will relieve him of scanning several displays and combining the information in some cognitive manner. This can be achieved by fusing sensor data into a unified display.

FIGURE 3
INCREASE IN NUMBER OF CONTROLS FOR VARIOUS HELICOPTERS



As an example of how a sensor fusion system might interface with the cockpit, let us look at the navigation task. This is one of the most difficult and time consuming tasks, particularly during NOE flight. Included in the overall long term navigation are many tasks:

- following a flight path
- making good estimated arrival times at specified waypoints
- assessing threats, bad weather, and obstacles along the flight path
- following terrain undulations
- planning alternate routes, and mission coordination with other force elements

If the pilot were to utilize dedicated displays to gather the data needed to perform these tasks, he would need at least four displays plus a hand-held map such as seen in Figure 4. These would include a map display, a radar altimeter, a display of enemy threats or obstacles, and a terrain imaging display such as visual observation or use of a FLIR, a radar or an optical image. Without a navigation system such as doppler, GPS, intertial, the pilot must continually assess his present position and his altitude to adjust his flight path to the desired flight path. He must simultaneously assess any threats and convert them to positions on his map or terrain displays. After he has done this, he must perform any rerouting of his course and altitude.

A computer controlled sensor fusion system could collect all relevant information and present it to the pilot in a single integrated display, thereby reducing his workload tremendously. Such a display might look like the one presented in Figure 5. Sensor fusion techniques would be employed to create the unified situational display of map data, threat data, flight plan data, present position and highlighted waypoints, contour data from both map and imaging sensors. Imaging data would provide changes to update the map features and permit accurate positioning. The display could also provide the desired flight path and could give alternate routes, perhaps as a function of threat position (entered from intelligence sources or sensed real time).

The display would be a true horizontal situation display providing all elements of the horizontal situation and some of the vertical situation (for example, color coding altitude, threat elevation, etc.).

The reduction of workload from such a combination of data will be of tremendous advantage. Other systems within the cockpit will utilize sensor fusion techniques to reduce even further the pilot workload.

6. FUSION IMPLEMENTATION

There are various levels of sensor/system fusion possible. Hardware fusion implies the sharing of common components such as apertures, optical elements, steering mechanisms, etc. A laser range finder integrated into a forward looking infrared system is one example. Display fusion, the type most commonly considered when discussing fusion, implies two or more information sources blended on a common display screen. A radar

FIGURE 4
CONVENTIONAL DISPLAYS USED DURING NAVIGATION TASKS

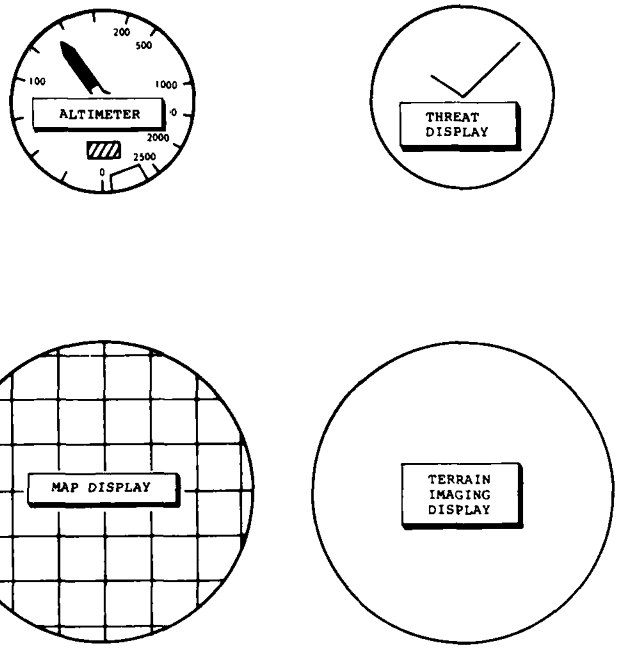
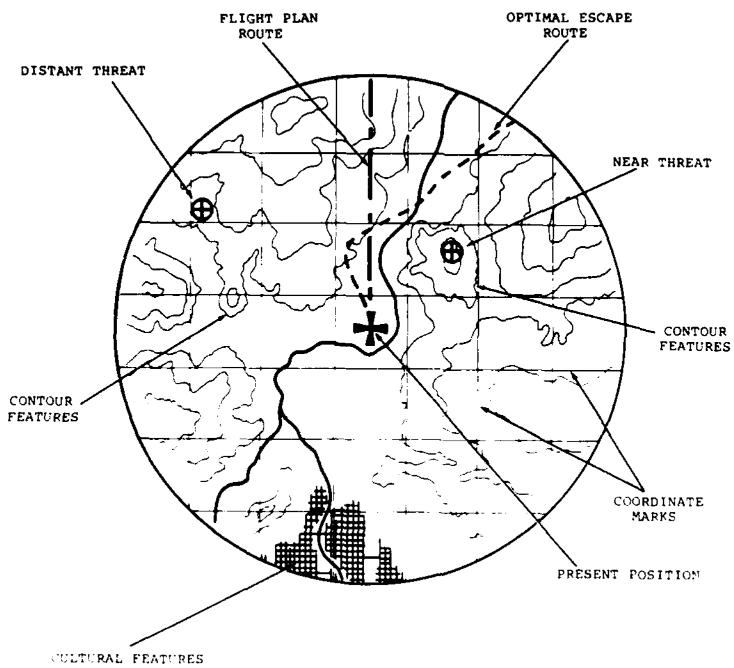


FIGURE 5
A POSSIBLE HORIZONTAL SITUATION DISPLAY
UTILIZING SENSOR FUSION INPUTS



"B" scan super-imposed on a digital terrain data base satisfies the criteria. Functional fusion accomplishes the task of assigning a task normally performed by one class of equipment to another. Passive ranging by a forward looking infrared sensor instead of active ranging by a radar is an example. The most interesting fusion hierarchy investigated is that of information fusion. Information fusion implies using the results of one or more processes to assist, modify, or initiate a different process.

Figure 6 illustrates a simplified version of this type of fusion. The hardware elements consist of a directional radar warning receiver, a forward looking infrared system, and a digital map driven by a precision navigation system. A threat emitter is detected by the radar warning receiver and the azimuth coordinates are transferred to the FLIR which then moves its field-of-view to the appropriate coordinates. The emitter is identified by its RF characteristics and the FLIR processor searches for that shape signature in the imagery. Simultaneously, the digital map data base is interrogated and probable location sites identified. These computed location sites are used by the FLIR processor to bias its decision algorithm, thereby increasing the detection probability and reducing the false alarm probability. Once the emitter has been identified by the FLIR, a passive ranging algorithm which uses the azimuth and elevation outputs from the FLIR in conjunction with the digital terrain data base computes the absolute location of the emitter and inserts that information into the terrain data base overlay. Meanwhile, the defensive suite on board has been alerted that some action is contemplated and the target may be handed to a radar homing missile while non-lethal defenses are readied. Concurrently, the FLIR processing system is programmed for launch plume identification and the threat is also handed off to the IR seeking missile along with its specific IR signature. The digital terrain data base may be interrogated and safe flight paths computed using intervening terrain features. Finally, the pilot is alerted to the threat, given reaction options and status of his defensive systems. This entire sequence could conceivably be accomplished in a sub-second interval. The key element to this information process is the cooperative interaction of typically independent subsystems.

This cooperative interaction implies certain system architecture attributes. Sensor data, which formerly was presented to the operator, must be converted to information that serves as the basis of future actions. Multiple, parallel processing paths must be available. Shared information buses or networks must connect all elements involved in the fusion process. Most importantly, the entire system must be adaptable based on mission scenario, threat, system performance degradation, and sensor information content. Management of this adaptable aspect is the most conceptually difficult of the many problems to be addressed in a fused system architecture.

Figure 7 illustrates a top level partitioning of the sensor/processor functions. Each sensor is considered as an information input port. The sensor operation mode is controlled ultimately by the processing system which may interleave various modes based on the mission management algorithm. For example, a radar pulse repetition frequency, wave form, transmit power and search azimuth/elevation may be functions of aircraft flight altitude, velocity, terrain conditions or other pertinent measured variables supplied by other sensors. The sensor transducer is considered a generalized analog measuring device with a known bandwidth, dynamic range and other characteristics of that particular measurement domain. The signal conditioner performs filtering, gain and other analog signal enhancement functions. The output of the signal conditioning function, which previously would be available for operator inspection, is fed to three different functions. The resident feature processor performs the non-linear analog operations in the time domain. The output of the feature processor determines which and what amount of data is to be communicated to the processing system. The second function fed is a data digitizer which transfers the digitized raw data to the formatter where the data is organized for transmission over the sensor bus network. These data are also available to the resident feature processor and the system state processor. The system state processor monitors the general sensor subsystem operation providing information on performance degradation, inter-sensor mode and other housekeeping information. The aircraft interface mechanism may be a gimbal or other mechanical connection to the aircraft.

Remembering that many sensor designs are predicated on the idea that the information gathered will be presented directly to the operator, new approaches should be investigated. For example, forward looking infrared systems scan at the rate of thirty to sixty hertz to eliminate the flicker at the display, a human oriented phenomena. Similarly, the human eye integration capability is used to enhance displayed signal-to-noise. This results in a tremendous redundancy of data frame to frame that may not be desirable for a fused processing system. Innovative approaches to gathering data in each measurement domain may have equal impact on system feasibility as future high speed processing hardware.

The operative element of the fused architecture is in the processing organization. Two main data streams are shown: the sensor data stream and the auxiliary control data stream. The sensor data stream consists of the output of the various sensing ports. These data are communicated over one or more high speed data buses where they are automatically routed to programmable processors. These processors reduce the data to information using such processing algorithms as auto-target recognition, doppler processing, frequency sorting, etc... Each processor is programmable and each is available for processing data from any sensor subsystem. For example, if the radar is not activated because of stealth consideration, the processing resources that would be used for radar can now be dedicated to passive ranging algorithms using the forward looking infrared as the data source. The new wrinkle in the sensor data stream processing chain is the effector processing element. Here the information fusion is achieved. Based on the information content and source, various automatic or semi-automatic actions can be implemented. Here artificial intelligence techniques come to bear using production rule or expert systems algorithms to proceed down an activity tree to the appropriate task.

The auxiliary data stream is operated upon in a similar manner. The control structure modifies the various modes of sensor operation under control of the executive program. This processing chain is also programmable and adaptable based upon the particular

FIGURE 6
SIMPLIFIED OPERATION EXAMPLE

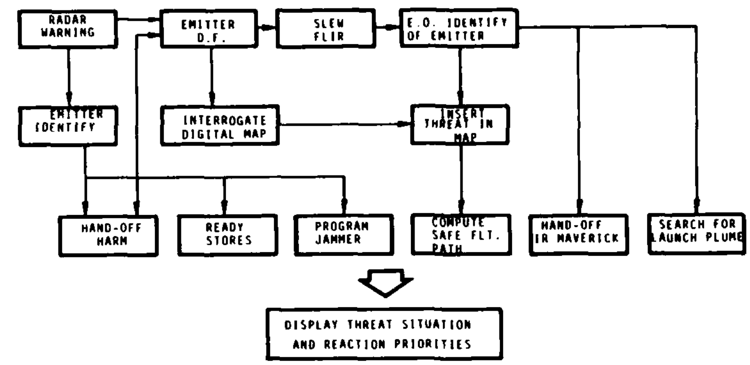


FIGURE 7
SENSOR FUNCTIONAL BLOCK DIAGRAM

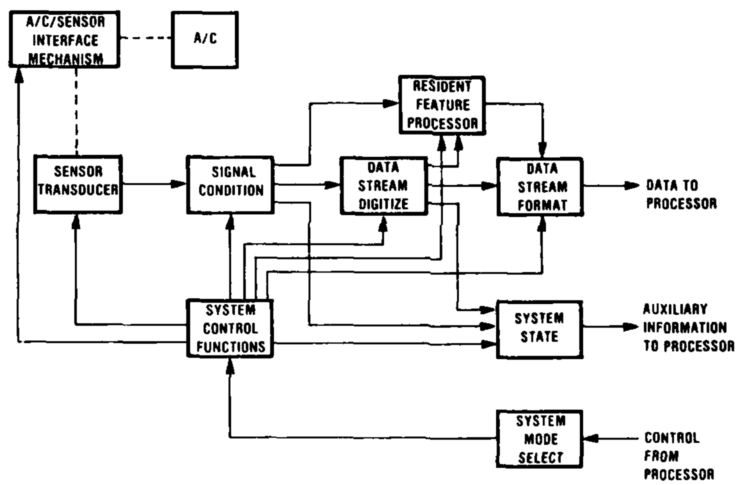
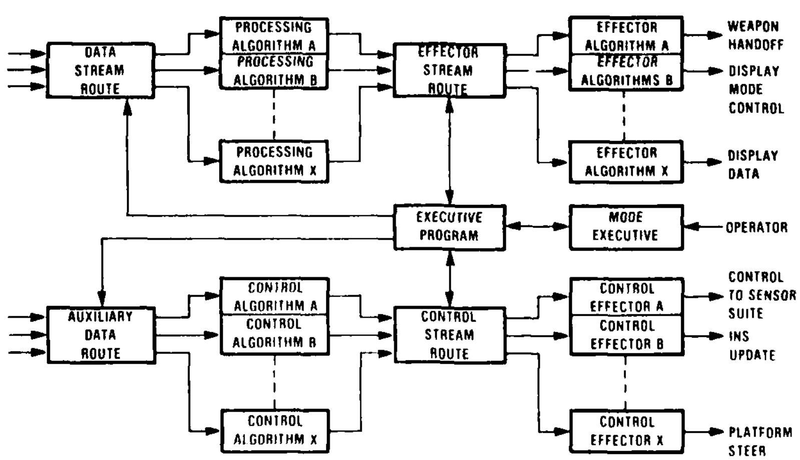


FIGURE 7 (CONTINUED)
SYSTEM FUNCTIONAL BLOCK DIAGRAM



situation. The system operator interacts through the mode executive modifying system overall operation and degree of authority desired.

Technologists usually refer to performance as the driving force in system design. Performance improvements such as extended range, greater coverage, improved sensitivity, etc. occupy many man years of research and design activity. Unfortunately, most of the mature technologies have essentially reached the point of diminishing return. The laws of physics establish the performance characteristics of most avionic equipments that will be available in the foreseeable future. A factor of ten more radar transmitting power may become available but that buys less than a factor of two more range. An infinite number of detectors on an infrared receiver focal plane will not defeat the smearing effect of diffraction. There will be performance improvements, but they will be wrested from nature at an exponentially increasing cost. What is needed is better ways of using the information currently available.

The user of advanced systems, while sensitive to performance, wants improved reliability, greater availability, less maintenance and, of course, lower cost. Increased complexity has become a burden instead of a boon to the user. Complexity does not necessarily result in negative system user qualities. The comparison of the mechanical calculator with today's scientific calculator is a good example. Which is more complex? The electronic calculator has orders of magnitude more "parts" but they are integrated in such a manner as to provide performance impossible with its mechanical counterpart while offering ease of use, reliability and at a cost that makes it a throw-away item when the battery finally fails. Not only that, the electronic calculator is technologically transparent to the user. That is, changes in semiconductor components, computational algorithms or circuit organization do not alter the method of using the instrument. Rapid evolution over the past decade has not negatively impacted the user except to provide improved capability at a cost to performance ratio much less than one. System fusion is merely the macro application of this micro concept.

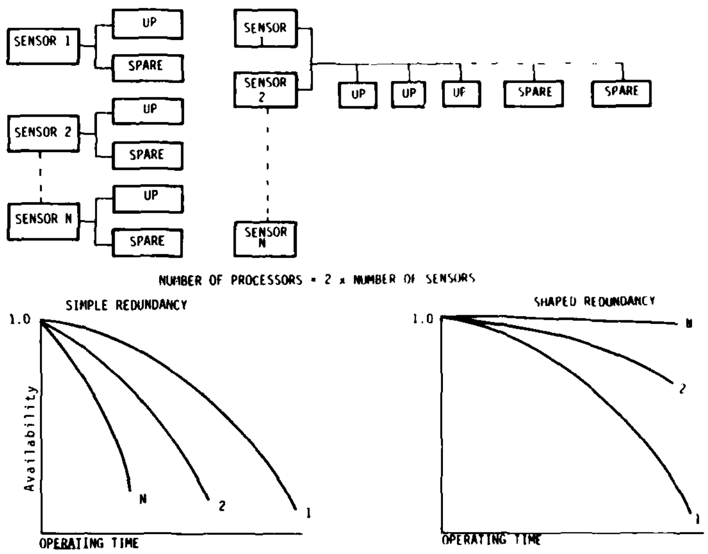
System fusion, executed intelligently, can provide the positive user qualities mentioned earlier and simultaneously increase overall system performance that must escape us if we attempt to work on one technology at a time. Figure 8 illustrates the reliability impact of fusion. Here, a system consisting of N sensing elements is integrated in two different ways. The system on the left is organized in the conventional manner with each sensing subsystem connected to its dedicated processor with one redundant processor to improve reliability. The fused system on the right is organized with the same N sensors connected to N processor nodes and N redundant processing nodes. Therefore, both systems have N sensors and 2xN processors. Assuming that all N sensor systems must be operational for mission success and that system complexity is a monotonically increasing function of N, then probability of mission performance as a function of operating time of just the processors is shown below as system complexity increases. The conventional system architecture yields a decreasing probability of mission success as complexity increases while the fused system yields greater and greater reliability as complexity increases. This is because the conventional system is only doubly redundant while the fused system has N redundancy in processors.

This simplified example obviously is not the entire story. Sensor bus, power supply, etc. failures are not addressed,but they too are benefited by the fused architecture. A sensor failure releases its processor to perform other tasks such as functional fusion as mentioned earlier. Navigation using terrain contours instead of a failed inertial reference unit is such as example. The conventional system organization provides no functional redundancy since its processing resources are unavailable for operation on foreign data.

In summary, the fused system is more reliable, less costly, lighter, and has more performance potential. New technological developments will not obsolete the fused system, merely obsolete the software dedicated to the old technology. System fusion provides a way to unload the harried system operator and assure mission success at a cost/performance ratio that decreases proportionally with overall system complexity.

Methods of implementing the fusion concept are under development. The convergence of high speed processing components, wide bandwidth buses, artificial intelligence techniques, advanced software and languages, along with a myriad of other digital technologies, makes system fusion a realistic prospect. In fact, the technological hurdles may be lower than the cultural ones. Management, engineering and procurement functions typically are not organized to implement such a concept. Where does the radar end and the FLIR begin? Who is responsible for system integration? From whom do you buy a fused avionic system?

FIGURE 8
ARCHITECTURE IMPACT ON SYSTEM AVAILABILITY



A REVIEW OF U.S. ARMY AIRCREW-AIRCRAFT INTEGRATION RESEARCH PROGRAMS

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SUMMARY

Handling qualities have historically been studied in the context of two-crew helicopters by stability and control engineers. Mission management development has been left to engineering psychologists or human factors specialists who have studied cockpit controls and displays independently. The desire of the Army for a one-crew helicopter that can perform the Scout and Attack role is forcing us to integrate these disciplines and concerns. This paper reviews some recent studies and results in these disciplines, describes the need for a more unified approach to support new helicopter development, and describes a plan to develop fundamental principles needed for efficient man-machine interface design.

1. INTRODUCTION

The primary task of the pilot of a two-crew helicopter is to fly the helicopter, that is, to perform the flightpath management function. The co-pilot's responsibilities include most of the other functions: navigation, communication, aircraft systems monitoring, and, in the military role, concern over threats, targets, and battle captain functions of command and control; these responsibilities will be defined as the mission management function. If the Army's desire to develop a one-crew version of the Light Helicopter Family (LHX) helicopter is to be realized, both flightpath management and mission management will have to be performed by one crew. This single-crew requirement means that flightpath control, that is, stability and control and handling qualities, must be studied in the context of the pilot being burdened with mission management tasks, and mission management needs to be studied in the context of a realistic flightpath management task. Historically, handling qualities have been studied by stability and control engineers with no duties other than flightpath control being required of the evaluation pilot. Mission management development has been left to engineering psychologists or to human factors specialists who have studied cockpit controls and displays independently. The desire of the Army for a one-crew helicopter that can perform the Scout and Attack role (LHX-SCAT) makes mandatory the integration of these disciplines and concerns (Fig. 1).

Working under the auspices of the Army/NASA Joint Agreement, the Army Aeromechanics Laboratory and NASA Ames Research Center have been pursuing both of these topics: handling qualities and human factors. This paper reviews some of the studies and results from the individual program elements; first, the stability and control and handling qualities, or flightpath management topics, and second, the human factors or mission management work. The final section of this paper describes the need for a more unified approach to support the LHX development and a plan for a new initiative to develop fundamental principles which are needed for efficient man-machine interface design.

2. FLIGHTPATH MANAGEMENT

The ability of a rotorcraft pilot to perform the flightpath management function is determined by the handling qualities of the vehicle: "those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot is able to perform the tasks required in support of an aircraft role" (Ref. 1). Handling qualities are determined not only by the stability and control characteristics of the vehicle, but also by the displays and controls which define the pilot-vehicle interface, the environmental characteristics, and the performance requirements for the task (Fig. 2).

The analysis of the effects of rotorcraft handling qualities on mission effectiveness is broken down into two components: (1) a determination of the influence of handling qualities parameters on the performance of the pilot-vehicle combination and on the physical and mental workload of the pilot, and (2) an analysis of the effects of the achieved precision of flightpath control and workload capacity of the pilot on selected measures of mission effectiveness. Handling qualities investigations by both NASA Ames and Army Aeromechanics Laboratory researchers have concentrated on the former component; these experiments have focused on nap-of-the-earth (NOE) mission tasks conducted during daytime or night/adverse weather conditions by a two-crew aircraft in which the pilot is only required to perform the flightpath management function. These programs have investigated either generic handling qualities effects or the handling qualities characteristics of specific rotorcraft configurations; the results of both types of programs are being used as sources of data upon which a revision to the U.S. military helicopter handling qualities specification, MIL-H-8501A, can be based (Ref. 2).

This section summarizes the results of NOE handling qualities investigations, for both day and night/adverse weather conditions, and describes an initial effort to relate achieved system performance and pilot workload to mission effectiveness.

2.1 NOE Flight Under Visual Meteorological Conditions

An initial series of helicopter handling qualities studies — including analysis, piloted simulation, and flight research (Table 1) — was conducted to assess the effects of rotor design parameters, interaxis coupling, and various levels of stability and control augmentation (Ref. 3). As a result, recommendations were made for: (1) minimum levels of pitch and roll damping and sensitivity; (2) maximum values of pitch-roll, collective-to-pitch, and collective-to-yaw coupling; and (3) generic stability and control augmentation system (SCAS) requirements.

The effects of thrust-response characteristics on helicopter handling qualities have, until recently, remained largely undefined. Helicopter thrust is influenced by several factors, including (1) engine governor dynamics, (2) vertical damping resulting from rotor inflow, and (3) the energy stored in the rotor, which is a function of rotor inertia. A multiphase program is being conducted to study these effects on helicopter handling qualities in hover and in representative low-speed NOE operations. To date, three moving-based piloted simulations (Refs. 4 and 5) have been conducted on the Vertical Motion Simulator (VMS) at Ames (Fig. 3). It was found that variations in the engine governor response time can have a significant effect on helicopter handling qualities. For the tasks evaluated, satisfactory handling qualities and rpm control were achieved only with a highly responsive governor, but increases in rotor inertia (thus in the stored kinetic energy) have only a minor, though desirable, effect on handling qualities (Fig. 4). The excess power requirement (T/W) was found to be a strong function of Z_w and is minimized at a Z_w-value around -0.8 rad/sec. The effect on handling qualities of requirements for pilot monitoring and control of rotor rpm can be significant. For a slow engine governor, the degradation in pilot rating in the bob-up tasks was as much as two ratings (Fig. 5). Techniques to relieve the pilot of the task and concern for monitoring proper rpm therefore need to be considered.

In support of the U.S. Army's Advanced Digital/Optical Control System (ADOCS) program, a series of piloted simulations was conducted both at the Boeing Vertol facility and on the VMS to assess the interactive effects of side-stick controller (SSC) characteristics and stability and control augmentation on handling qualities. An initial experiment (Ref. 6) revealed that angular rate stabilization in pitch and roll was sufficient to provide satisfactory handling qualities when a two-axis SSC was employed for control of those axes; however, when a rigid three- or four-axis device (which added directional and directional-plus-collective control, respectively, to the SSC) was employed, attitude stabilization was required to maintain adequate handling qualities. These results were substantiated and expanded upon by the Ref. 7 experiment which demonstrated that a four-axis, small-deflection SSC yielded satisfactory handling qualities for NOE tasks when integrated with a SCAS that incorporated higher levels of augmentation; however, separated controllers (Fig. 6) were required to maintain satisfactory handling qualities for the more demanding control tasks or when reduced levels of stability and control augmentation were provided.

Current research programs being conducted to support the development of the handling qualities specification include investigations of roll-control requirements, hover and low-speed directional control characteristics, and helicopter air combat maneuverability and agility requirements.

A major shortcoming in the current handling qualities data base is known to be roll-control effectiveness. This critical and fundamental criterion can have a major effect on the basic design of a helicopter. Analyses and piloted simulations are being conducted to assess required levels of damping and the control power required to trim, to recover from external upsets, and to maneuver for various rotorcraft configurations operating in an NOE environment. Similarly, to compensate for a lack of mission-oriented handling qualities data, a piloted simulation is being conducted to evaluate the effects of: (1) mission task requirements; (2) basic yaw sensitivity and damping; (3) directional gust sensitivity; and (4) yaw SCAS implementation on the handling qualities of generic-LHX candidates, including tilt-rotor, coaxial rotor, and no-tail-rotor configurations (Fig. 7).

To support the requirement for an air-to-air combat capability for future military helicopters, a facility is being developed which can be used to investigate handling qualities requirements in terrain flight air combat. One-on-one air combat (Fig. 8) is simulated using the VMS as the cockpit of the friendly aircraft which is engaged in a computer-generated visual data base by an enemy aircraft which may be flown manually from a fixed-base station or automatically through an interactive maneuvering algorithm. Variations in the performance, stability and control, controllers, and displays of the friendly aircraft are being investigated.

2.2 Effects of Night/Adverse Weather Conditions

The requirement that military rotorcraft operations be conducted at night and under other conditions of limited visibility has given impetus to research programs designed to investigate the interactive effects of vision aids and displays on NOE handling qualities.

In a program conducted to support the development of the Advanced Attack Helicopter (AAH), various levels of stability and control augmentation together with variations in the format and dynamics of the symbols provided on the Pilot Night Vision System (PNVS) (Fig. 9) were investigated in a piloted simulation (Ref. 8). It was found that the handling qualities of the baseline control/display system were unsatisfactory without improvement; recommendations for alterations to the PNVS symbol dynamics and the implementation of a velocity-command system for a hover/bob-up/weapon delivery task were made to the Army Program Manager.

An investigation involving the simulation of a less complex night vision aid was carried out to support the Army Helicopter Improvement Program (AHIP) (Ref. 9). In this simulation, the effects of presenting the PNVS flight symbology on a panel-mounted display (PMD) versus a head-up display (HUD) were compared for a nighttime scout helicopter mission in which the pilot was provided with simulated night vision goggles. Although no clear preference for the HUD or PMD was established, the use of the display improved handling qualities for the lower levels of augmentation. However, higher levels of augmentation, which

included a velocity-command system and augmentation of the directional and vertical axes, were required for satisfactory handling qualities.

The state-of-the-art night vision system for combat helicopters includes a visually coupled helmet-mounted display of infrared imagery and superimposed symbology: the Integrated Helmet and Display Sight System (IHADSS) (Fig. 10). This system was employed in two simulator investigations (Refs. 10 and 11) designed to assess the effects of reduced visibility conditions on the ADOCS visual flight simulation results cited previously. Significant degradations in handling qualities occurred for most tasks flown with the IHADSS relative to the identical tasks flown under visual flight conditions (Fig. 11). In general, higher levels of stability augmentation were required to achieve handling qualities comparable to those achieved for the visual flight tasks.

2.3 Handling Qualities Effects on Mission Effectiveness

A preliminary computer simulation was conducted to relate certain handling qualities effects, such as precision of flightpath control and pilot workload, to the ability of a single scout helicopter, or helicopter team, to accomplish a specified anti-armor mission successfully (Ref. 12). A key feature of the program is a simulation of microterrain features and their effects on detection, exposure, and masking for NOE flight.

For the purpose of this study, degraded scout helicopter handling qualities were assumed to manifest themselves in four ways: (1) increases in the basic NOE altitude at which the helicopter can fly at a given speed, (2) increases in the amount and frequency content of altitude excursions above the basic NOE altitude, (3) increases in the amount and frequency content of altitude excursions in hover above that required for observation, and (4) decreases in the amount of visual free time available to the crew for surveillance and fire control functions. The effects of each of these parameters on selected measures of effectiveness (MOE) were investigated separately for three different combat scenarios. These MOE included primary measures such as: (1) the probability of the scout(s) being killed: $P_K(B)$, (2) the number of enemy vehicles killed: $N_K(R)$, and (3) the exchange ratio: number of enemy vehicles killed divided by the number of scouts killed (E/R). Certain intermediate MOE, involving detection probabilities and average times required to detect and kill, were also analyzed to gain further insight into the engagement outcomes.

In order to assess the overall effect of handling qualities on the MOE, three "grades" of handling qualities — "perfect," "fair," and "bad" — were defined by specifying the associated values of basic NOE altitude, NOE altitude error, hover altitude error, and visual free time. The resultant values of the primary MOE for each grade of handling qualities are presented in Fig. 12.

This study demonstrated that handling qualities do have a significant effect on the ability to perform a specific mission, as indicated by variations in the selected MOE. This effect resulted primarily from variations in the probability of the scout helicopter being detected, particularly during a precision hover.

3. MISSION MANAGEMENT

The objectives of the mission management or human factors part of the program are: (1) to explore and develop the fundamental principles and methodologies necessary to exploit pilot perceptual, motor, and information processing capacity for application to advanced helicopter cockpit design, and (2) to develop objective and predictive techniques for assessing pilot workload.

One of the first experimental efforts under this program to address the pilot control/display interface was to determine the relative location of flight displays and the corresponding controls. Specifically, the altimeter and rate of climb indicators which are conventionally located to the right of the pilot centerline indicate parameters which are controlled by the collective stick in the left hand. The flexibility of new electronic display formats such as the PNVS (see Section 2) afforded a reasonable opportunity to determine if there was any penalty caused by this opposite or contralateral control display relationship. In the previously mentioned PNVS study (Ref. 13) most of the pilots preferred a same side (ipsilateral) arrangement. An experiment was conducted to test for any measurable differences in the time to effect control when the display is contralateral rather than ipsilateral to the controller.

Performance was assessed based on reaction time and total time to null the error. An index of difficulty for performing the task was hypothesized based on Fitts law (Ref. 14) which indicates that the time to effect a reduction in error amplitude, A, to a given target with width, W, varies in a linear fashion with the index of difficulty defined as $ID = A + B \log_2(2A/W)$. Thus, the total performance time could be plotted versus ID as shown in Fig. 13. A surprising result is the difference in slopes of the two cases, since this implies that an increase in the difficulty of task propagates into the control phase.

To investigate the same question in a more realistic situation, the experiment was repeated with both collective and cyclic controls being used to null errors simultaneously (Ref 15). In this experiment the rudder pedals also required attention to null a randomly disturbed heading reference symbol. The display was as shown in Fig. 14. For the ipsilateral case, the V- and H-scales were interchanged. The index of difficulty for the two-axis task was redefined by simply summing the ID defined by the previous equation for the two components of the task. Subjective reports indicated that this definition did not reflect the actual difficulty of the dual task. For example, combinations with both targets on the same side of center (up or down) were easier to capture than targets with the same ID having targets on opposite sides of center, and any combination that included one wide target was easier than a combination with the same ID but containing a narrower target. These findings suggest that more work will be required to establish a meaningful index of difficulty for two-axis tasks.

Other interesting results are illustrated by Fig. 15 which shows that the first response and first capture of target were faster with the contralateral configuration, but the second response and second (final) capture were faster with the ipsilateral configuration. This result suggests that the contralateral display makes it difficult for the subjects to develop a strategy for both controls so that they react to one target at a time, thus initiating the movement sooner (initial response) but taking longer to complete the task (final capture) since the final capture requires manipulating both controls simultaneously. If this analysis is correct, the question as to whether or not the traditional contralateral control display configuration is the most efficient for helicopters depends on whether a configuration that encourages a segmented processing and movement strategy is better than one that elicits a more integrated natural response. This question has not yet been addressed.

Another study on cockpit flight controls was performed under contract by Sikorsky Aircraft Division (Ref. 16). The experiment investigated the use of multi-axis sidestick controls for flightpath control in configurations such as were developed for the ADOCS program and the simultaneous performance of a keyboard entry task with the free hand. As would be expected, the results show that keyboard entry tasks interfere with the performance of flightpath tracking, and, conversely, the flightpath tracking interfered with keyboard entry. If a degradation in performance occurs, the use of a multi-axis controller to free a hand for mission management tasks may not be appropriate. The ADOCS data (Section 2) generally show that for most tasks, with a high level of SCAS, similar pilot ratings can be obtained independently of the level of controller integration. However, as the SCAS degrades, separated controls generally become superior. This result has implications on reliability which must be designed into the flight control system SCAS; the four-axis controller may imply a mission-critical SCAS, or even a flight-critical SCAS at more complex levels. This requirement may force the costs associated with a fully integrated controller to a prohibitively high level.

An alternative approach, which provides the ability to change control and display functions without removing the hand from the flight controls or directing visual attention to switch or function locations, would be attractive in an NOE environment and is a logical situation in which to incorporate voice command and display technology.

3.1 Voice Command and Display (SCADS)

It has recently become technologically feasible for the pilot to control onboard systems by voice command, and to receive feedback on this control process via synthesized speech. Research has been performed at Ames on both of these aspects for several years. The helicopter environment makes the accomplishment of accurate automatic speech recognition difficult because of the noise and vibration, as well as physiological and psychological factors such as stress, fear, and fatigue. However, studies have shown encouragingly high accuracy rates.

Development of speech output principles has also been pursued for several years, and an example of applying these concepts is the voice interactive electronic warning system (VIEWS) research project conducted at the Aeromechanics Laboratory.

This study (Ref. 17) was conducted at the request of the Aircraft Survivability Equipment (ASE) Program Manager (PM) and was designed to examine the use of an integrated visual and speech display for a threat warning system. The current Radar Warning Receiver (APR-39) uses a combination of visual strobe lines and proportional rate frequency audio (PRF) tones to give pilots information concerning the location of enemy radar emitters. The PM requested assistance in defining a set of visual symbols to replace the strobe lines, and a set of voice messages to replace the PRF tones. Integrated displays create a new set of problems not found in visual or speech displays alone. The two most apparent problems are display priority and temporal veridicality.

<u>Display priority</u> — Visual displays can display more than one item of information simultaneously; speech displays can only present one item of information at a time. Most visual displays do not attempt to prioritize information; this task is left to the pilot. Speech displays must prioritize information output if they are to be effective. This system prioritization can ease the decisionmaking task of the pilot, but this requires higher levels of "intelligence" on the part of the system.

<u>Temporal veridicality</u> — Visual displays, because of their instantaneous nature can change rapidly to always give veridical information; speech systems, because of the time required to articulate a message may lag behind actual events, particularly if the messages are stored and delivered as strings of words. Integrated visual and speech displays may therefore give conflicting information and cause a pilot to lose confidence in the system.

The following are some of the points which resulted from the VIEWS project:

1. The prioritization logic eliminated all message cueing and updated each word just prior to it being spoken. It also implemented a message update called a "coda" at the end of a message that has been spoken while the real time situation was changing. This coda eliminated the need to repeat a whole message to give an up-to-date output.

2. A special symbol (message being spoken pointer) was displayed on the visual display screen directly under the visual symbol that the speech message was addressing. Thus the pilot always knew which visual symbol the speech display was talking about.

3. It was determined through testing that pilots could use either the visual or speech systems to successfully avoid radar guided threats, but they preferred to have both systems working together.

3.2 Pilot Workload Assessment

Several approaches towards assessing pilot workload have been proposed. According to a study by Phatak (Ref. 18) these methods fall into the following general categories:

1. Methods based upon secondary task performance.

2. Physiological measurement methods.

3. Methods based upon primary task performance.

4. Method using subjective opinion rating/scale.

5. Time line and task analysis methods.

6. Pilot model methods.

The secondary task performance method has the possibility of the secondary task affecting or modifying the pilot's performance and/or strategy in accomplishing the primary task. A popular secondary task method that has been applied to handling qualities work (Ref. 19) is the Sternberg task where the pilot is given several letters to remember, then asked to decide if a letter presented at a certain frequency during the test is in or out of his group. The study (Ref. 20) by Hemingway applied this technique during a related helicopter handling qualities study. For several reasons, including the methodology, no clear correlations were obtained.

The use of physiological measures of the operator for assessing workload is restricted because physiological metrics only measure states of arousal and do not represent measures of pilot workload except under special situations.

Closed-loop system performance on the primary task is generally not a satisfactory measure of workload because of its relative insensitivity to large variations in workload except at the extremely low or high levels.

A pilot's evaluation or opinion about a task provides the most direct window into the mental perception, or notion, of experienced workload. However, even this approach is fraught with methodological problems related to standardization of terminology and the large degree of intra- and inter-subject variability in the subjective interpretation of the factors perceived to be contributing to workload. In spite of these drawbacks, the bottom line in the acceptance of any new system is the pilot's subjective opinion or assessment of the system performance and required workload.

Time line analysis methods are based upon the intuitive notion that workload must be related to the time pressure imposed upon the human operator performing a given task. These methods use systematic task analysis procedures to estimate the time needed to complete each elemental or primitive task and hence the total time required for accomplishing the overall task. One problem, of course, is that some tasks are very much more difficult to perform than other tasks even though they perhaps take the same amount of time.

None of the above methods provides the system designer significant insight into identifying the individual factors or components of human effort which are responsible for the increased pilot workload. Furthermore, the measures may only be used to assess the pilot workload for existing systems and are not suitable for workload prediction in the design phase of building a new system.

A much better understanding of the fundamental issues embodied in the concept of workload may be possible with models that describe the perceptual, cognitive, and motor processes actually used by the human pilot in accomplishing a given task. The use of mathematical modeling as a tool for analyzing man-systems performance has been of substantial interest to researchers for over 30 years. During that period the human has been characterized as a servo-compensator, a sample data controller, a finite-state machine, an optimal controller, and most recently as an intelligent system. Although there is currently no clear consensus about the utility of available model-based methods for assessing pilot workload and performance in realistic military helicopter missions, the potential benefits are such that we have a continuing effort to develop such models.

3.3 Expert Systems and Artificial Intelligence

With the need to simplify the total pilot workload, there is impetus to help with decisionmaking and to automate certain tasks. A grant with the Ohio State University is addressing the question of the cost and benefit of one crew and high automation versus two crew and nominal automation. The approach is an iterative program of experimental studies using a video game-like task followed by an analytical effort employing discrete control modeling. The goal of this effort is to produce a predictive methodology to aid in the understanding of human supervisory control of highly interactive systems. In addition, a contract has been initiated with Perceptronics, Inc., to use the modified Petri-net as an analytical tool for developing guidelines and concept designs for incorporating artifical intelligence and smart systems techniques into LHX cockpit automation features.

4. AIRCREW-AIRCRAFT INTEGRATION PLANS

Except for single pilot IFR in the civil/FAA context, single-crew concepts have not been considered in helicopter flight control research. If the tasks performed by the co-pilot are to be taken over by the pilot, increased levels of automation are required. The LHX will need control laws for automatic and manual control of flightpath including integration with propulsion, fire control, and navigation functions. Configuration effects such as thrust vectoring and X-force control will also have to be taken into account if the LHX configurations is a compound helicopter, ABC configuration, or a tilt rotor. In addition, concepts for safety-of-flight automation will have to be developed for such functions as obstacle avoidance, threat avoidance, flight-envelope limiting, and automatic failure recovery.

These developments will have to rely heavily on ground-based simulation and will require high-fidelity dynamic simulation such as will be available in the Rotorcraft Systems Integration Simulator (RSIS) (Ref. 21) at Ames Research Center. In addition, to adequately represent the pilot's mission-management functions such as battle captain tasks, navigation, and aircraft systems management, it will be necessary to develop surrogate tasks which can be incorporated in the simulation on a realistic real-time basis; the cognitive workload associated with battle management may have a significant impact on total mission performance and realistic simulation of these functions is considered particularly important.

Numerous LHX Man-Machine issues remain as unknowns. The extremely difficult task of flying NOE at night and in weather will leave the pilot little capacity to perform his battle management functions unless extensive innovation is applied to all the man-machine interface tasks. The allocation of control and display media between manual, visual and voice, the extent of automation, and the application of artificial intelligence/expert systems will have to be extensive, yet little is known to guide the appropriate choice of these applications.

For the night and poor weather situations, candidate external scene visual displays which will permit single-crew operation for the LHX mission tasks must be assessed. Wide field-of-view display devices are in the embryonic stage even for ground-based simulators; other display devices, such as night vision goggles, HUD, and IHADSS, have not been applied to such a demanding role. Sensor fusion and real-time image processing for both flight and target tasks have not been developed for an operational system. Not only are hardware advances needed, but a better knowledge of the required functional capabilities, such as field-of-view, resolution, detail, and image update rates, must also be developed to guide the hardware design objectives.

In addition to the outside world visual scene, it will be necessary to display to the pilot an easily understood image of the tactical situation and navigation functions. The achievement of this capability will require the development of real-time tactical situation scenarios which can be used to investigate the man-machine interface required for battle captain functions such as target engagement and threat defense.

Artificial intelligence and expert systems will be required to aid the pilots' decisionmaking tasks and to automate routine prescribed functions. Replacement or supplementation of specific manual controls and visual displays with speech recognition and speech generation techniques is intuitively appealing for pilot workload reduction. However, a significant amount of work will be required to determine which functions are best controlled by voice, how these voice modes should be implemented, and how they are to be interfaced with other modes. Finally, a better understanding is required on how a human interacts with a highly automated system so that the dynamics of switching from one automated mode to another, or back to a manual function as the mission needs change can be defined, and so that guidelines can be developed for the synthesis of the total cockpit.

Some of the problems described above will be addressed in the Advanced Rotorcraft Technology Integration program (Ref. 22) and these results will form the basis for the LHX cockpit design. In addition, the work described in Sections 2 and 3 will be expanded to improve understanding of the fundamental questions. In recognition of a lack of a fundamental approach to the pilot-cockpit design, a new initiative has been developed and will be initiated towards the end of FY 1984.

5. ARMY/NASA AIRCREW-AIRCRAFT INTEGRATION PROGRAM

The objective of this joint Army/NASA program is a focused effort to develop a validated predictive methodology: a set of analytic structures with which cost-effective and efficient guidelines and principles for man-machine integration designs can be derived before a commitment to hardware is made. The analytic (modeling) approach is motivated by the high cost of redesign and retrofit of nonoptimal systems and the ever-increasing cost of the training simulators and systems required to support the operational units in the field. The focus of the program will be the mission of a single-crew scout/attack helicopter operating at night, in adverse weather, in the NOE environment. Although the aircraft will employ the most advanced technology, this mission will produce extreme workload, demand superior performance, and require extensive training of the aircrew. The essential issues are the triad of pilot workload, performance, and training which are inexorably intertwined and affect all integrated design considerations in future helicopter cockpits. Current design practice relies on a cut-and-try approach, and on questionable procedures for evaluating effectiveness. Consequently, it is not possible to quantify what is essential to the design of a system for an effective man-machine interface and, therefore, there exist no future benefits from lessons learned.

To achieve the objective, a fundamental understanding must be established of how the human operator processes the information by which he perceives his environment, how he acts upon that perception, how training modified this perception, and how the foregoing relate to pilot performance and workload. Considerable research has already been accomplished in an attempt to understand human perception and cognition and to establish measures of pilot performance and workload. These efforts have generally been ad hoc and fragmented, the results have seldom been focused on the design of a man-machine system and have never been conveyed in terms useful to the engineering user community.

The planned program will be an interdisciplinary effort involving pilots, display engineers, control engineers, mathematicians, and engineering psychologists. Essential tools for this program will be flexible, versatile, ground-based, and in-flight simulator research capabilities that permit the study of the interactions of variations in display laws and control laws on the human's ability to interface with automatic aids in order to perform specified missions. The ground-based simulation capability at Ames is already exceptional and will be augmented when the Rotorcraft Systems Integration Simulator and NASA's Manned Vehicle Research Simulation Facility are put into operation. The in-flight research capability could, for example, be provided by an integration in the UH-60A Black Hawk of the ADOCS flight controls

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SUPPORT HELD AT M..(U) ADVISORY GROUP FOR AEROSPACE
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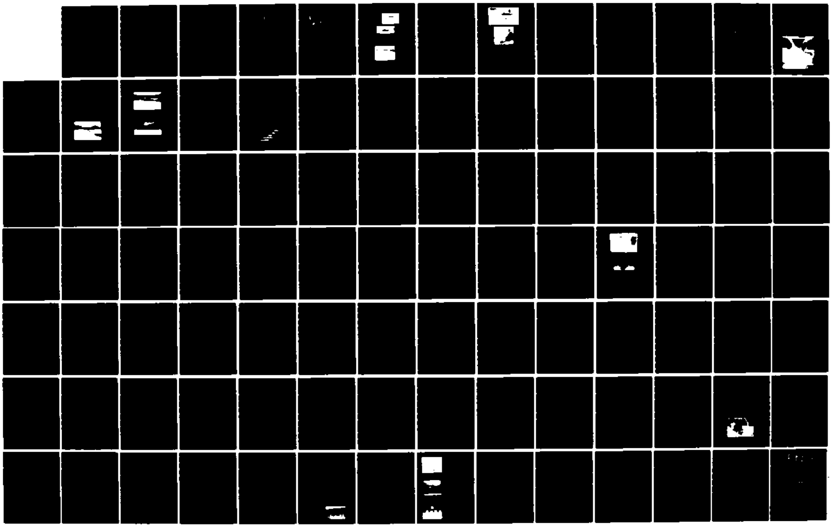

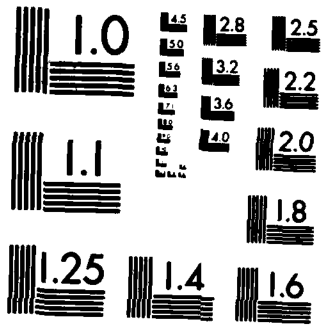

MICROCOPY RESOLUTION TEST CHART
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and NASA/Army digital avionics packages. Inhouse efforts utilizing these unique facilities will be designed to complement contracted work.

The program will consist of seven phases (Fig. 16). A program schedule is shown in Figure 17.

6. CONCLUSIONS

Handling qualities research conducted by the U.S. Army Aeromechanics Laboratory and NASA Ames Research Center to date has emphasized the interactive effects of basic stability and control characteristics, type of SCAS, controller characteristics, and vision aids and displays on the ability of a two-crew rotorcraft to conduct specific NOE mission tasks. Extrapolation to the single-crew situation from these data must be based on sound engineering and piloting judgment.

Numerous single-crew helicopter man-machine issues remain as unknowns. The extremely difficult task of flying NOE at night and in weather will leave the pilot little capacity to perform his battle management functions unless extensive innovation is applied to all the man-machine interface tasks. The allocation of control and display media between manual, visual and voice, the extent of automation, and the application of artificial intelligence/expert systems will have to be extensive, yet little is known to guide the appropriate choice of these applications. To address these concerns a new program is planned which will be an interdisciplinary effort involving pilots, display engineers, control engineers, mathematicians, and engineering psychologists. The objective of this joint Army/NASA program is predictive methodology, a set of analytic structures with which cost-effective and efficient guidelines and principles for man-machine integration designs can be derived before a commitment to hardware is made.

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TABLE 1. SUMMARY OF INITIAL TERRAIN FLIGHT EXPERIMENTS

Experiments	Objective	Tasks	Simulator	Rotor type	Control system type
I	To determine effect of large variations in rotor design parameters	Longitudinal vertical task Lateral slalom task Combined task	Fixed base (Ames S-19)	Teetering Articulated Hingeless	Basic helicopter (rate-type in pitch, roll, and yaw)
II	To assess effect of various levels of SCAS	Combined task	Moving base (Ames FSAA)	Teetering Articulated Hingeless	SCAS Input Decoupling Rate command Attitude command in pitch and roll
III	To evaluate a sophisticated SCAS for hingeless rotor helicopter	Combined task	Moving base (Ames FSAA)	Hingeless	SCAS Attitude and rate Stability augmentation Control augmentation
IV	To investigate roll damping, roll sensitivity, and pitch-roll cross-coupling and correlate results with Experiments I and II	Prescribed lateral slalom course over a runway	In-flight (UH-1H/VSTOLAND)	Teetering	Rate-type in pitch, roll, and yaw

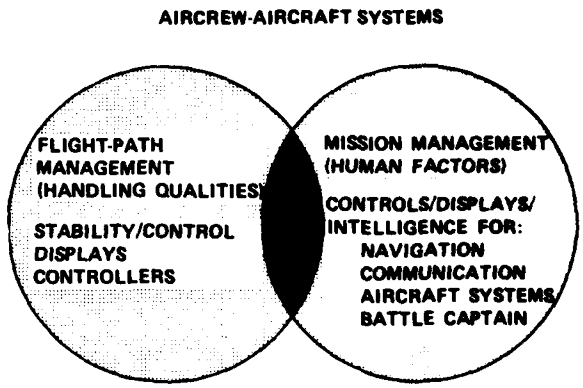


Fig. 1. Flightpath/mission management interaction.

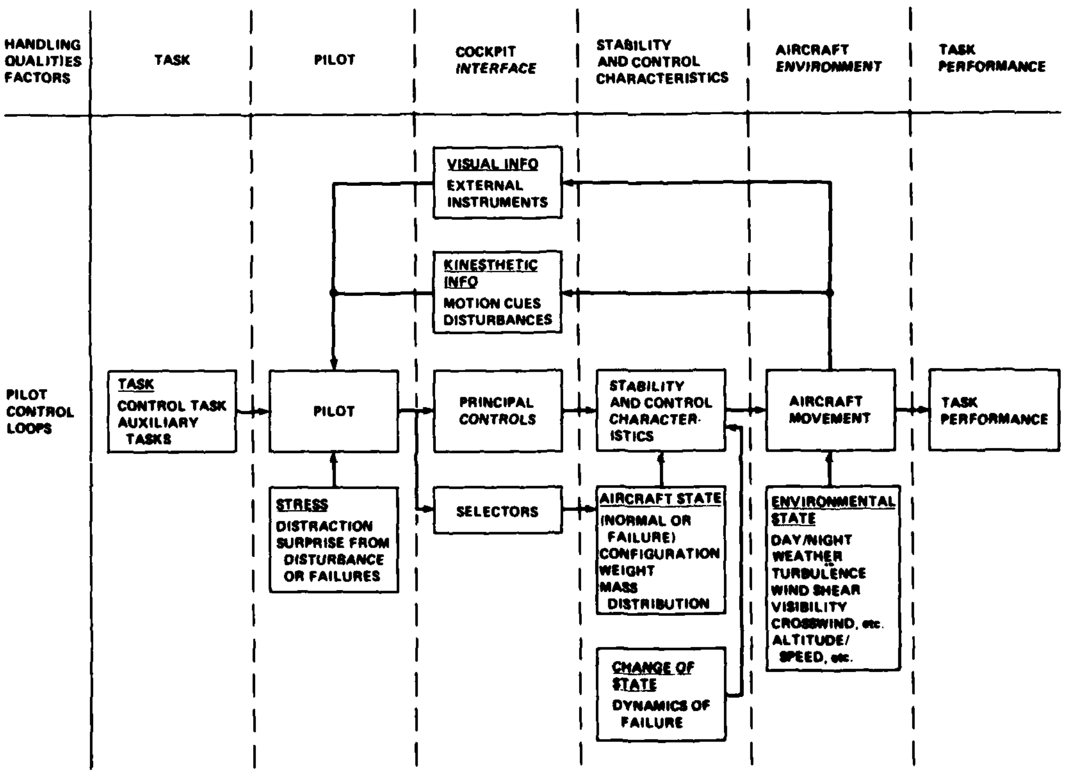


Fig. 2. Elements of control loop that influence handling qualities (Ref. 1).

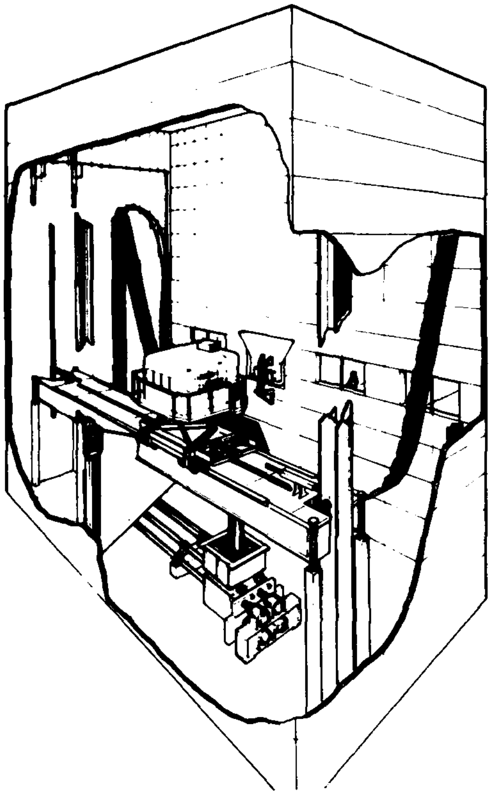


Fig. 3. NASA Ames Vertical Motion Simulator (VMS).

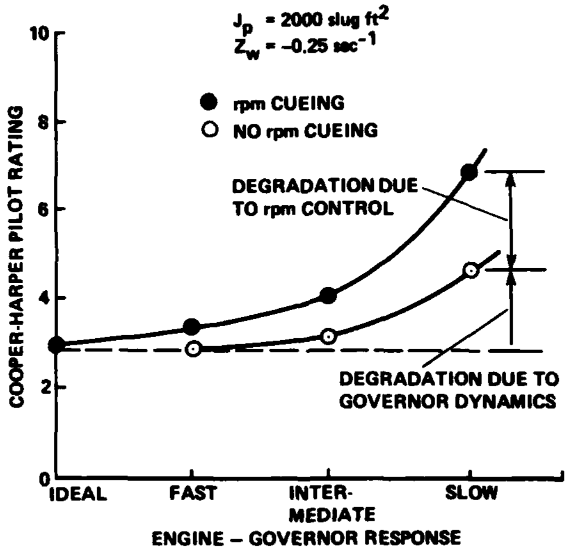


Fig. 5. Effect of requiring rpm control.

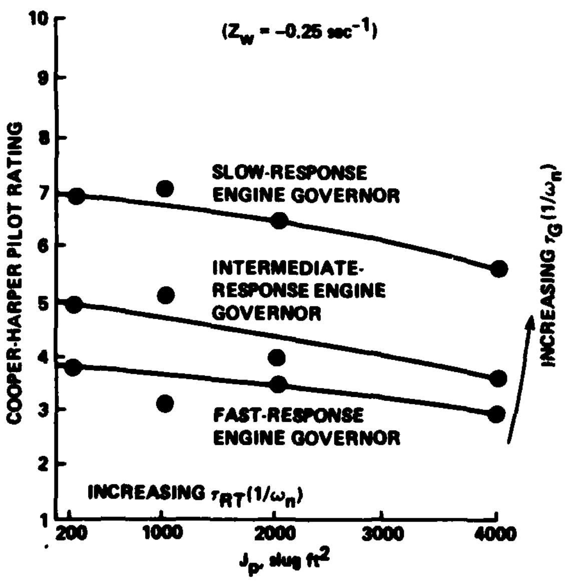


Fig. 4. Effect of rotor inertia and engine governor.

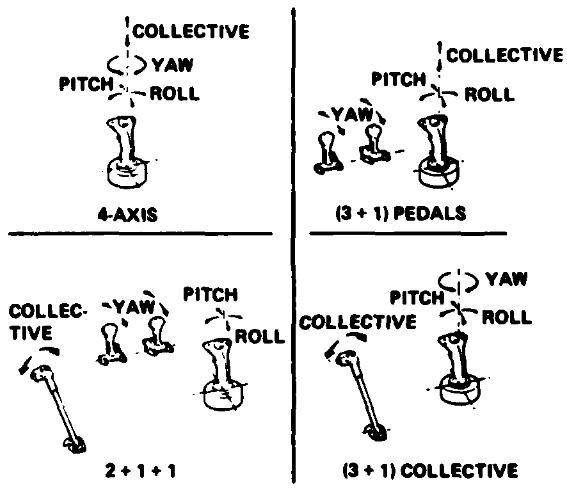


Fig. 6. Controller configurations.

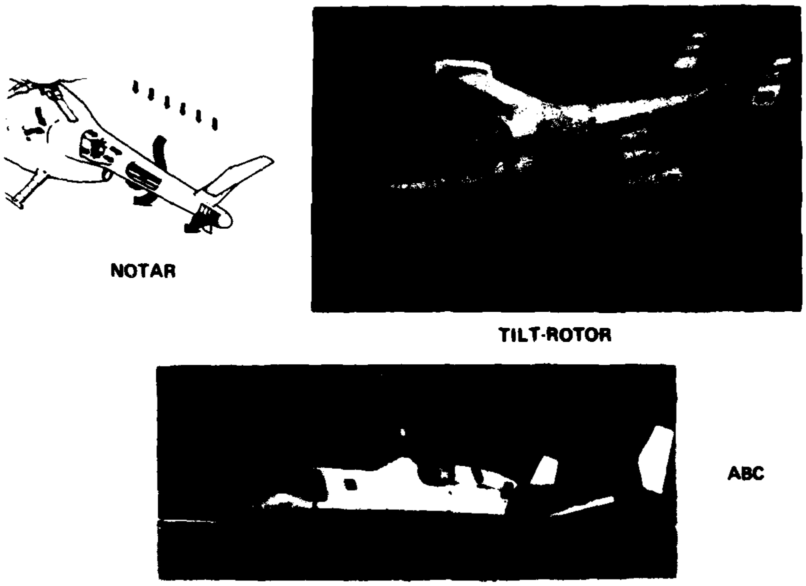


Fig. 7. Generic LHX-configurations.

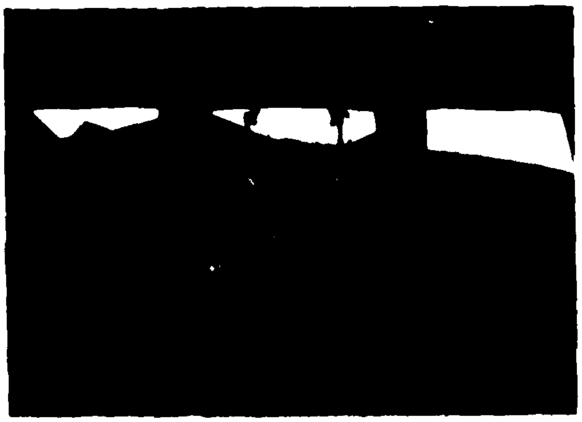


Fig. 8. Simulation of air-to-air combat.

SYMBOL	INFORMATION
1. AIRCRAFT REFERENCE	FIXED REFERENCE FOR HORIZON LINE, VELOCITY VECTOR, HOVER POSITION, CYCLIC DIRECTOR, AND FIRE CONTROL SYMBOLS
2. HORIZON LINE (CRUISE MODE ONLY)	PITCH AND ROLL ATTITUDE WITH RESPECT TO AIRCRAFT REFERENCE (INDICATING NOSE-UP PITCH AND LEFT ROLL)
3. VELOCITY VECTOR	HORIZONTAL DOPPLER VELOCITY COMPONENTS (INDICATING FORWARD AND RIGHT DRIFT VELOCITIES)
4. HOVER POSITION	DESIGNATED HOVER POSITION WITH RESPECT TO AIRCRAFT REFERENCE SYMBOL (INDICATING AIRCRAFT FORWARD AND TO RIGHT OF DESIRED HOVER POSITION)
5. CYCLIC DIRECTOR	CYCLIC STICK COMMAND WITH RESPECT TO HOVER POSITION SYMBOL (INDICATING LEFT AND AFT CYCLIC STICK REQUIRED TO RETURN TO DESIGNATED HOVER POSITION)

CENTRAL SYMBOLOGY

SYMBOL	INFORMATION
6. AIRCRAFT HEADING	MOVING TAPE INDICATION OF HEADING (INDICATING NORTH)
7. HEADING ERROR	HEADING AT TIME BOB-UP MODE SELECTED (INDICATING 030)
8. RADAR ALTITUDE	HEIGHT ABOVE GROUND LEVEL IN BOTH ANALOG AND DIGITAL FORM (INDICATING 50 ft)
9. RATE OF CLIMB	MOVING POINTER WITH FULL-SCALE DEFLECTION OF ±1,000 ft/min (INDICATING 0 ft/min)
10. LATERAL ACCELERATION	INCLINOMETER INDICATION OF SIDE FORCE
11. AIRSPEED	DIGITAL READOUT IN knots
12. TORQUE	ENGINE TORQUE IN percent

PERIPHERAL SYMBOLOGY

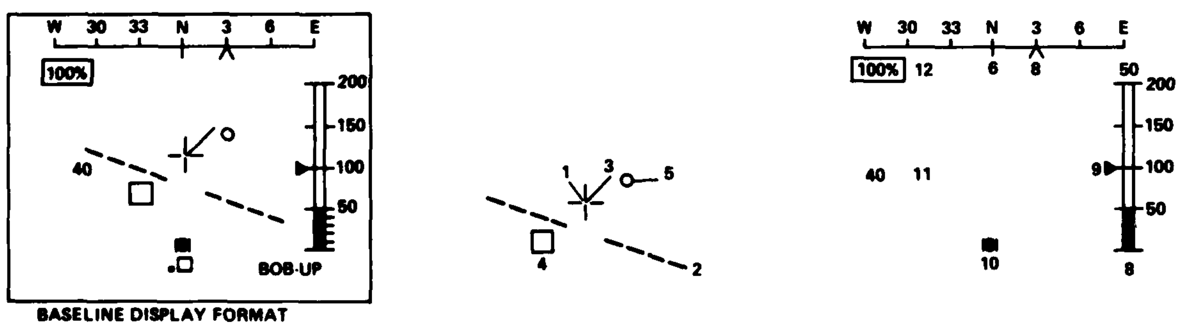
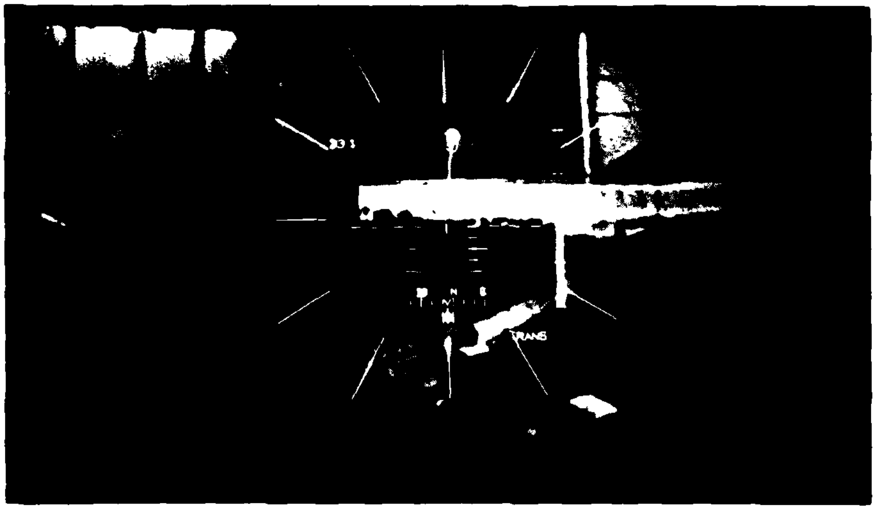


Fig. 9. PNS display mode symbology.



(a) SUPERIMPOSED SYMBOLS

(b) INTEGRATED HELMET AND DISPLAY SIGHT SYSTEM INSTALLATION

Fig. 10. Helmet-mounted display.

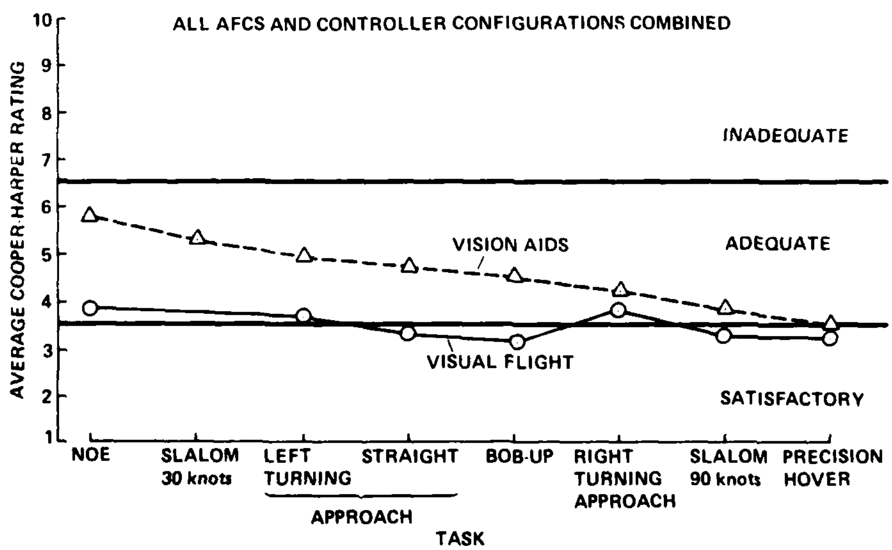


Fig. 11. Effect of reduced visibility conditions on pilot ratings.

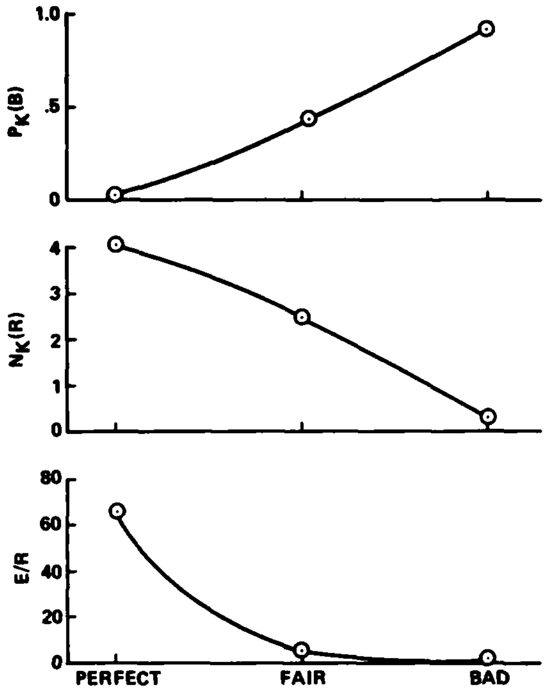


Fig. 12. Combined effect of handling qualities parameters.

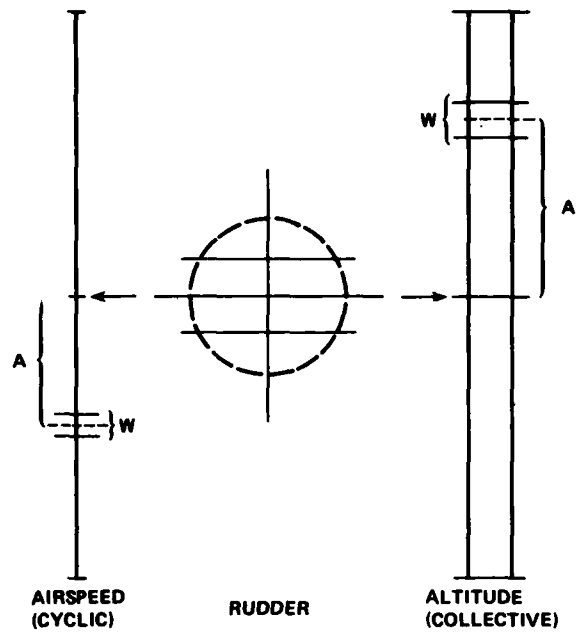


Fig. 14. Modified pilot night vision system (PNVS) display, contralateral configuration, showing amplitude (A) and width (W) of sample targets.

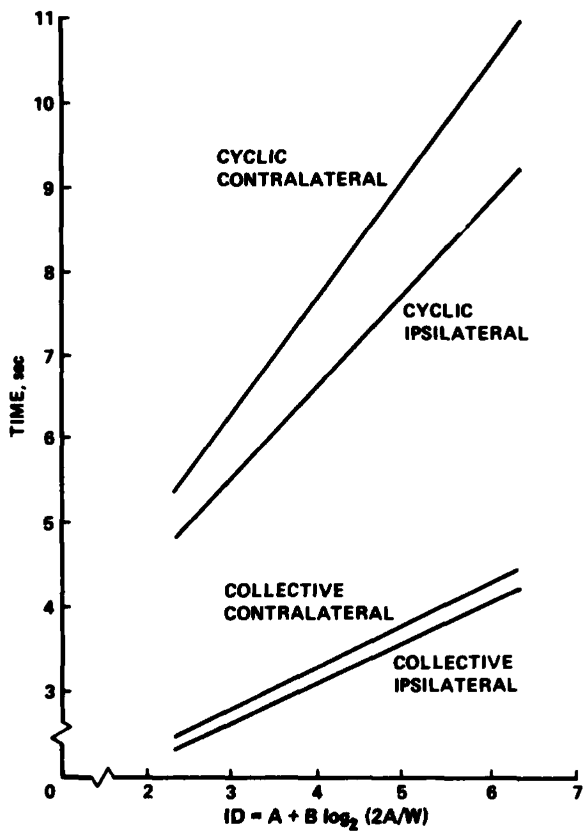


Fig. 13. Capture line variation with index of difficulty.

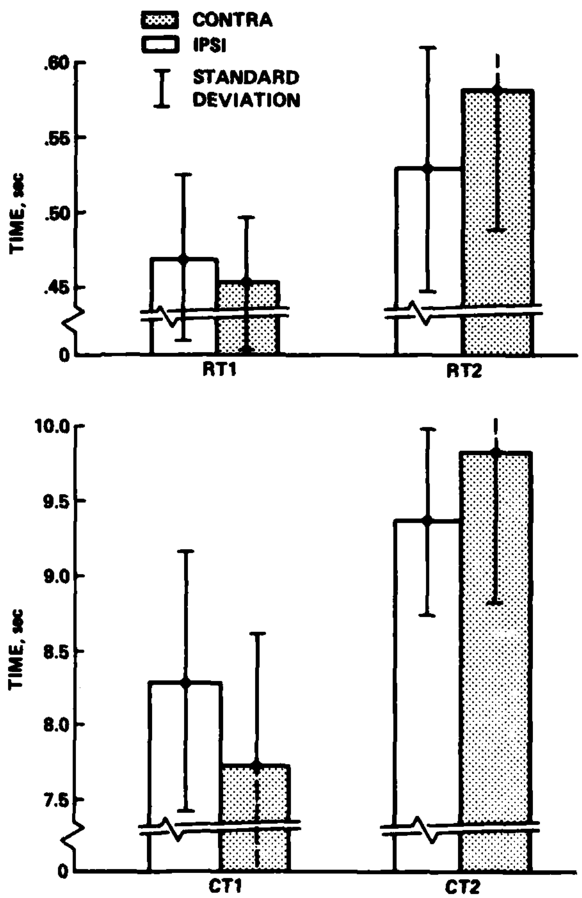


Fig. 15. Illustration of first and second response (RT1 and RT2), and first and second capture (CT1 and CT2).

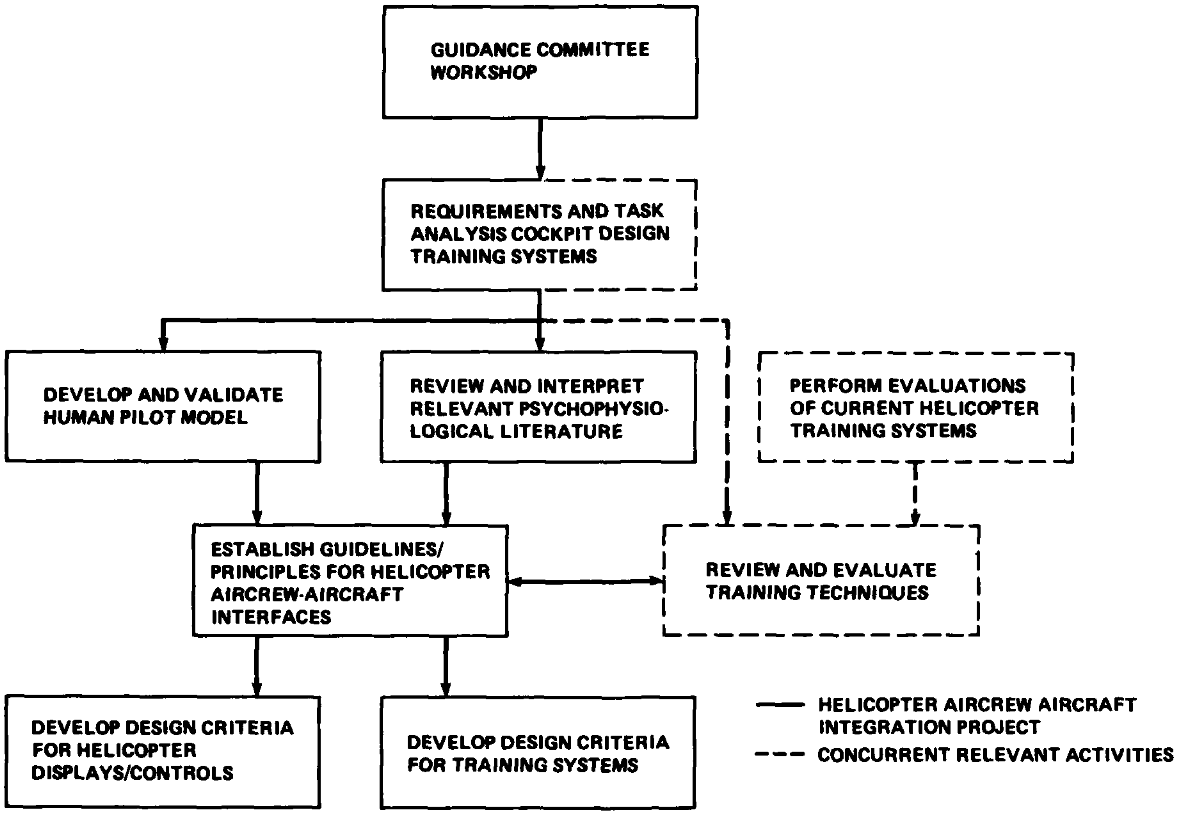


Fig. 16. Approach to aircrew-aircraft-integration project.

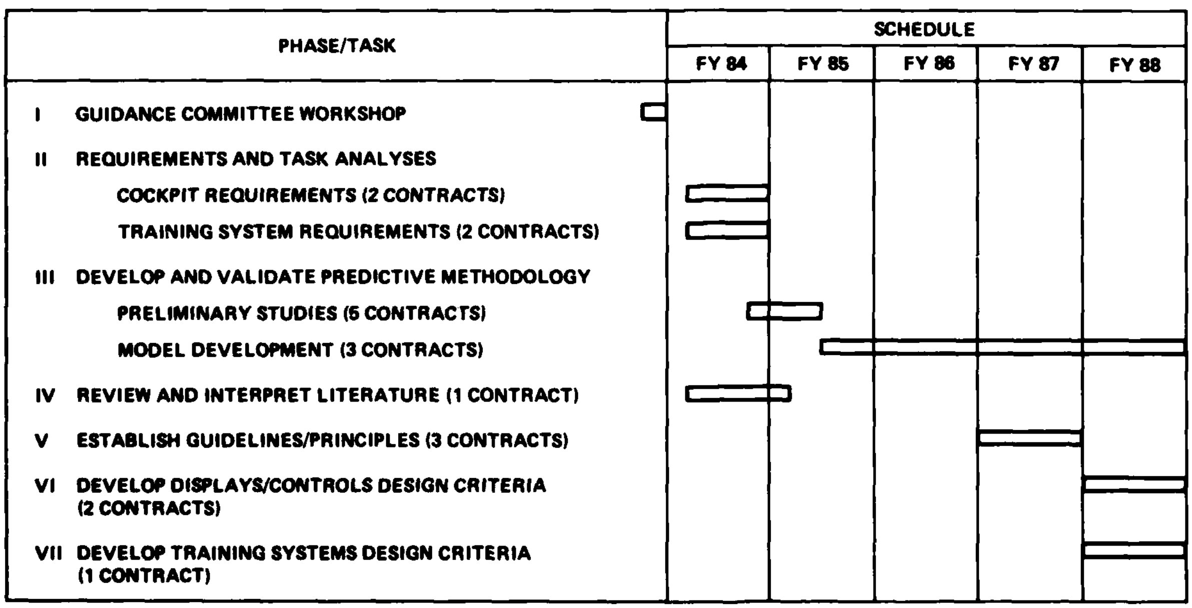


Fig. 17. Helicopter aircrew-aircraft-integration program schedule.

ADVANCED FLIGHT SIMULATION FOR HELICOPTER DEVELOPMENT

by

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SUMMARY

A new advanced simulation capability for both fixed-wing and rotary-wing aircraft has been developed at MBB. The fixed-based simulator consists of interchangeable cockpit stations, a computer generated imagery (CGI) visual system, both coupled with the math model simulation computer. Specific integration rigs for avionics, flight controls and weapon systems can be operated with the simulator.

The cockpit for helicopter applications, at the present stage, consists of a BO 105 cockpit including original flight controls, trim system, and standard instruments. The visual scene is produced by a computer generated image system (CGI), consisting of the data base, image generator, and projection unit. The system allows different visibility conditions and simulation of fixed and moving objects. The projection system presently consists of a three channel beam splitter, work is under way to install a spherical dome with a 5 channel projection system. A comprehensive helicopter math model is applied representing the aerodynamic and dynamic complexities of rotory-wing aircraft. Some model details are fully non-linear aerodynamics, rigid body and blade dynamics, individual blade modelling and engine/governor dynamics.

The paper gives a description of the simulation system including cockpit/interface hardware, visual display system and mathematical modelling. The simulation validity is demonstrated by the verification of the mathematical model and by pilot judgements. Typical simulation tasks for future military applications are discussed. It is concluded that the existing simulator provides a valid tool for helicopter system development.

1. INTRODUCTION

During the past few years, a rapidly increasing application of flight simulation during research, design and development work of helicopters has become evident. While flight simulation has been intensively used by fixed-wing designers for a much longer time, the helicopter industry has been more reserved toward piloted flight simulation applications during development work. There were manifold reasons for this attitude: Helicopter modelling, although showing continuous improvements in the important areas, was obviously not fully sufficient for achieving full acceptance in development simulations. Computer limitations were additionally restricting the complexity of mathematical models that could be executed in real time. The visual display quality was usually not adequate to fulfill the required high standards for helicopter simulation. Deficiencies in motion cueing were further major factors in the problem.

The changing attitude toward simulator application during helicopter development work results from different reasons: Firstly, design requirements for future helicopter systems are becoming more and more demanding; it is no longer sufficient, and may become extremely costly and time-consuming, to solve problems by a classic "cut-and-try" technique during flight testing. Secondly, there are major efforts to replace current helicopter flying qualities criteria, and to develop new specifications containing based mission-oriented handling qualities requirements (1). Special considerations will be given to mission flight phases and environmental conditions (day/night, visibility, terrain nature, atmospheric conditions). To consider these types of criteria, analytical technicques will be necessary which represent the rotorcraft and the pilot in a closed-loop. Finally, similarly as with military fixed-wing aircraft, future military helicopters will represent complex weapon systems. Development of these systems must not only concentrate on the basic vehicle optimization, but has to put the main emphasis on the integration of all interrelated elements, such as basic vehicle, flight controls, displays and vision aids, weapon system and the human element the pilot. Total system simulation techniques are required to handle the complex interrelation of the above elements. Reference (2) defines some of the requirements for adequate mission tasks simulations.

The studies have been sponsored in part by the Ministry of Defence of the Federal Republic of Germany

2. NEW MBB SIMULATION FACILITY

In addition to various simulation facilities at MBB, a program was imitiated by the MBB Military Aircraft Division to develop a new advanced simulation capability, Reference (3). An overview of the facility is shown in Figure 1. The total system simulator should permit many studies during both military fixed-wing aircraft and rotary wing aircraft research and development. To enable an easy change of simulation between these two types of air vehicles, a high degree of flexibility was one of the principal design goals. The following discussion of the facility is limited to those areas defining the specific application of the simulation for helicopter purposes.

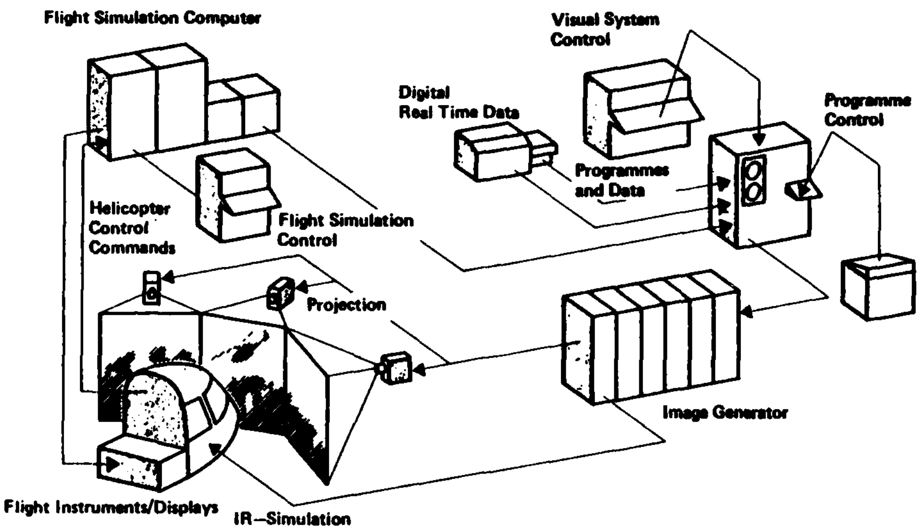

Figure 1 Development Simulator with Visual System

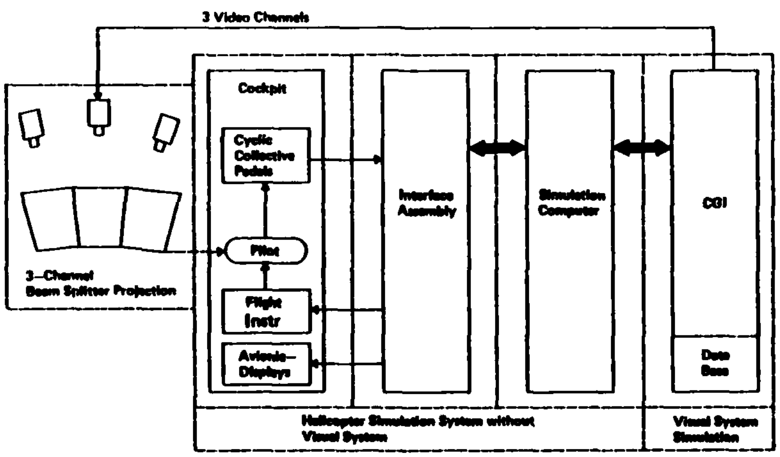

Figure 2 Functional Blockdiagram of the Helicopter Simulation System

Major elements of the system include (Figure 1):

- the fixed-based helicopter cockpit
- the cockpit interface
- the digital flight simulation computer
- the computer image generator
- and the visual projection system.

A functional block diagram of the helicopter simulation system is shown in Figure 2. The helicopter cockpit and the cockpit interface are systems specific to the helicopter simulation. Interchangeable cockpit stations and specific integration-rigs, e.g. for equipment and flight control systems can be operated with the simulator using suitable interfaces.

2.1 COCKPIT ASSEMBLY

The fixed-base cockpit presently available for helicopter simulations is an original BO 105 fuselage. The dual-seat cockpit is fitted with the original instrument panel including all basic flight and performance instruments necessary for flight operation, i.e. airspeed altimeter, artificial horizon, rotor rpm, and engine instruments. Figure 3 shows internal cockpit details and the external view of the CGI-scene. The original primary flying controls (collective lever, stick, pedals) are installed in the cockpit. A trim switch on the cyclic grip can be used as in the original rotorcraft, to operate a parallel actuator and a springbox which produces the artificial cyclic stick control forces. Collective lever friction forces are produced as original, and identical pedal forces are simulated by an electromagnetic friction brake. A central operating console allows the selection of the modes for the simulation and interface computers and to automatically adapt trim positions for the initial flight status. Pre-calculation of the desired trim conditions by an automatic trim process (6 degrees of freedom) enables start of the simulation at any desired intitial condition, a facility which proved to be highly helpful for development simulations.

The data link between the cockpit and the simulation computer is managed by an interface assembly. A functional block diagram of the unit is shown in Figure 4. The assembly converts and scales the analog data from the cockpit into the format of the simulation computer and vice versa. Power supplies for the cockpit are contained in and drive voltages for the electronics and the reference voltages are delivered by this assembly. Analog signals from the control levers are fed via analog/digital converters

Figure 3 Internal Cockpit Details and outside view

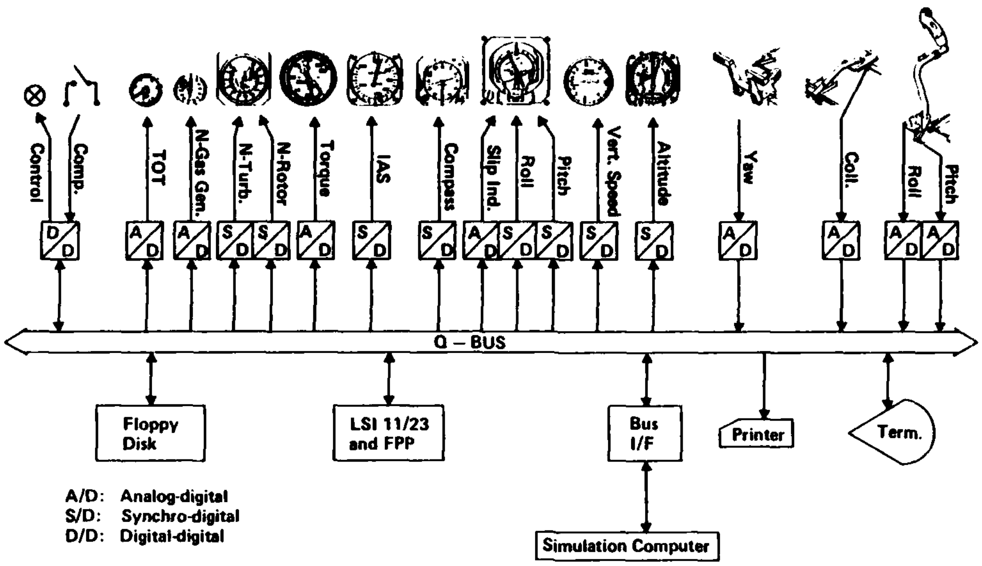

Figure 4 Blockdiagram of the Interface Assembly

to the interface. Data flow is managed by a parallel bus to and from the simulation computer, triggered by the interface computer. In addition, the interface assembly serves as a signal distribution unit allowing the recording and measurement of signals, analog and digital. To drive the conventional cockpit instruments, digital-synchros and digital-analog converters are available. A "quick-look" evaluation of the flight mechanics data is possible with an analog plotter.

With recent progress in visual system simulation (especially advanced CGI-systems), the need for complete motion system simulations was substantially reduced. For the existing simulator two systems were selected simulating the onset of motion and the vibratory content (Table of Figure 5). By inflating a cushion on the pilot's seat and backrest, translational and rotational accelerations are simulated, within a frequency range of 0 to 3 Hz. A buffeting system is installed, allowing simulation of vertical accelerations and vibrations up to ±2 g maximum within a total frequency range of 3 to 35 Hz.

• G-Seat	Inflatable cushion varies hardness and pressure distribution of the seat and backrest
	Frequency Range 0 to 3 Hz
• Buffet System	Vertical Acceleration and Vibration of the Cockpit
	Frequency Range 3 to 35 Hz
	Maximum Acceleration ±2 g

Figure 5 G-Load Simulation

2.2 VISUAL DISPLAY SYSTEM

The visual system simulation for a simulator is one of the factors defining the quality of simulation task performance. Many of the military rotorcraft operations are typically near hovering and low speed operations, in close proximity to the ground, with typical maneuvers consisting of terrain flying profiles, such as NOE, contour, and low level flight. Steep takeoff and landing profiles are further typical for helicopter operations. The requirements resulting from these flight conditions are especially challenging the selection and the design of the visual system simulation.

In the new MBB simulation facility a computer-generated imagery (CGI) visual system has been installed, offering the most promise, especially in the field-of-view. The scene generated displays contour images in correct perspective, real time and visually significant features of the real world environment. In the following sections some details of the system will be shortly described, including the digital image generation, the data base, and the projection system.

Digital Image Generator - The system design is based on a completely digital approach up to the provided output feeding the projection units. The TV-type pictures are synthetically generated form an offline produced numerical description of the landscape data base. Some important technical data are listed in Figure 6.

Number of Channels	**3 (Extension 5)**
Edges per Scene	**8000**
Field of View	**horizontal: 106 deg** **vertical: 23 deg**
Frame Rate	**30/sec**
Colours	**256**
Texturing	
Moving Targets	**3**
Simulation of Day Time	**Day, Night, Dusk**
Brightness	**6 Foot Lamhort**
Resolution	**1 m rad**

Figure 6 Technical Data of Visual System Simulation

The basis of the image generating system is a General-Electric CGI-Compu-Scene II. The system allows the generation of 8000 edges maximum per scene, which can be divided into five channels simultaneously for projection. The system presently operating uses only 3 channels. The calculation of images in real time apply appropriate transformation processes in the digitally stored landscape, to produce a true perspective scene. The system features special effects like curve shading, texturing and smoothing of edges, which substantially improve the perspective impression without the use of the limited number of edges. The system allows simulation of moving models (helicopters, tanks), controlled by an operator or by a computer. Various visibility conditions can be simulated by changing the color of each face. The system, thereby, allows simulation of variable weather conditions (dusk, fog), and day and night conditions.

Figure 7 Typical Runway CGI-Scene

Data Base - At the moment, as data base, various digitized areas originally created for fixed wing applications are used. Figure 7 shows an airfield area in Germany, including the runway, buildings and wide sorroundings. Since the visual cues representation for detailed helicopter NOE flying simulation requires more scene details on the ground, the data are presently being extended. Areas of high detail are created by introducing houses, trees or other visual elements from a special model library. In addition, the data base offers the representation of geographical attributes like hills, rivers, lakes, routes etc. Figure 8 illustrates a detailed CGI-scene, including typical available

elements. To ease pilots assessment how close the helicopter is flying over the ground, a textured relief showing gradual differences is laid over the surface. An impression of the available technique of achieving high quality of CGI-scene is given in Figure 9. The picture represents a single channel projection, containing a maximum of 8000 edges in the scene. The helicopter shown is a typical representation of the moving targets.

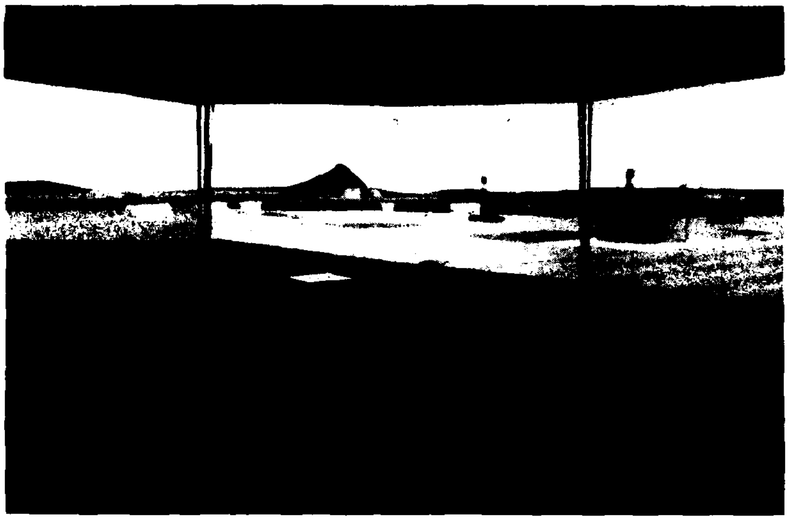

Figure 8 Detailed CGI-Scene

Figure 9 CGI-Scene illustrating visual content and Textural Relief

Projection System - The projection system actually consists of three television monitors representing three visual channels. The system presently available can provide a total field of view of 106° in azimuth and 23° in elevation. The 3-channel beam splitter is installed in front of the interchangeable cockpits, with the three optical axes intersecting in the pilot's eye point. Figure 10 shows a cross-section of the beam geometry.

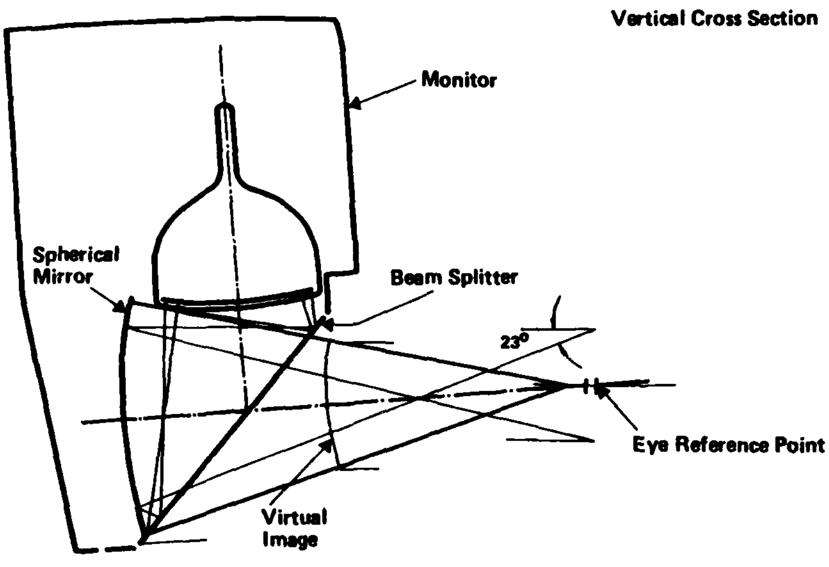

Figure 10 Beam Splitter Projection

For the next phase work is under way to install a spherical dome with a 5 channel projection system using 6 light valve projectors simultaneously. The sixth projector can be used alternately to cover an additional view angle as required. The principle build-up of the dome and the projection system is illustrated in Figure 11. With this final

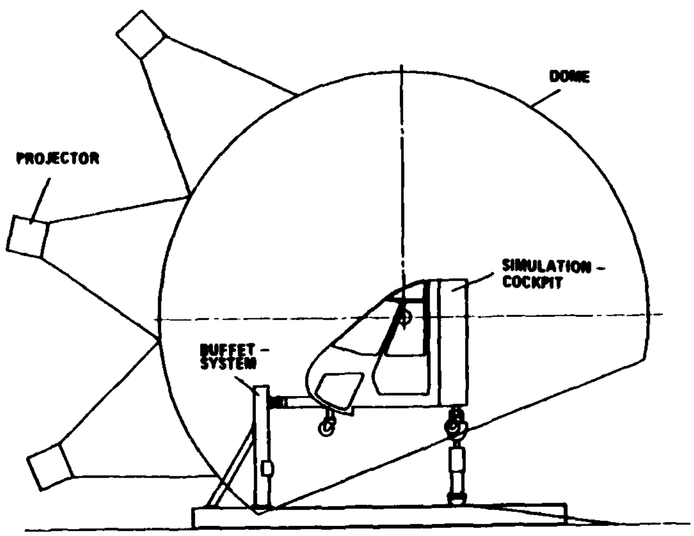

Figure 11 Projection Dome

stage the total field-of-view for helicopter simulations will cover an area of 180° in azimuth and 130° in elevation (a capability for a 300° by 150° is provided for later extensions). Figure 12 shows the coverage of the present 3 channel projection and of the increased field of the 5 channel system, compared with the pilot's outside visual field from a typical side-by-side cockpit. The comparison with the MIL-STD 850 B requirement is also shown.

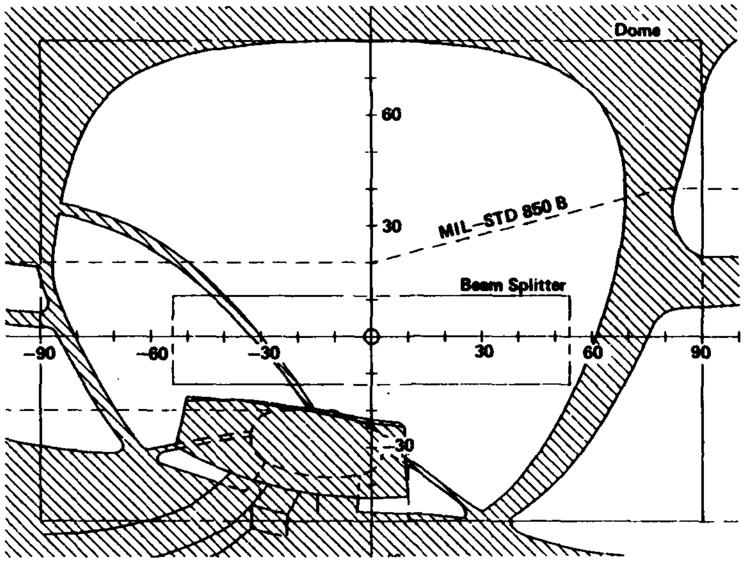

Figure 12 Field of View for a Side by Side Cockpit (Pilot's Position)

3. MATHEMATICAL MODEL

3.1 MODEL BUILD-UP

The requirements on the mathematical modelling for advanced helicopter simulations have dramatically increased, since studies for future systems must necessarily be aimed at the exploration of specific flying qualities within extended boundaries. The target in the development of the mathematical model for the simulator was also to maintain, as far as possible, the high standard of the existing flight mechanics model, as used in the unmanned simulation studies. The model applied is a comprehensive helicopter model of the total-force type. It represents the effects of non-linear aerodynamics, rigid body and rotor dynamics and considers all other components (fuselage, empennages, tail rotor) realistically.

Rotor Aerodynamics - The aerodynamic model is based on blade-element theory, including the effects of compressibility, stall, and reverse flow effects, obtained from experimental airfoil data. Rotor downwash is modelled by a modified momentum theory, with quasi-dynamic downwash effects being simulated by introducing a time constant in the induced velocity calculation. The influence on the rotor of the ground effect, which is important for simulations during low altitude flight is modelled by a downwash reduction as a function of the rotor to ground separation.

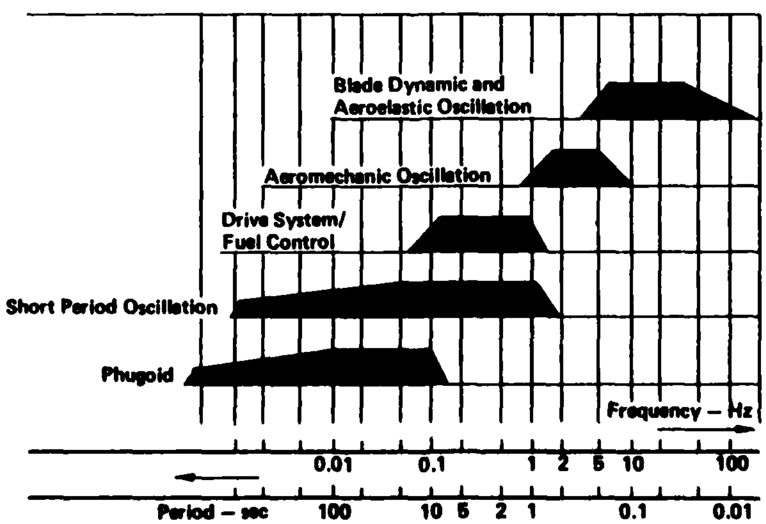

Figure 13 Characteristic Frequencies of the Helicopter

Rotor Dynamics - The necessary bandwidth for modelling rotor dynamics is determined by the extent to which, for example, aeroelastic effects or high-gain feed-back control systems are to be investigated. The frequency range of interest is summarized in Figure 13 which shows the typical ranges of body modes, engine and rotor modes being close together. To adequately model the rotor and body modes coupling, the rotor is represented by an individual blade model, considering flapping on each of the actual blades separately Reference (4). The principle of the individual blade calculation is shown in Figure 14. Different integration step lengths are applied for the blades and rigid body motion. This type of rotor model results in a demand for extensive computing capacity, which is made possible by the parallel processor architecture of the Denelcor, Inc. HEP (Heterogenous Element Processor) simulation computer, in which currently up to ten parallel processes can be performed (10^7 - 10^8 MIPS, 1δ2 x 10^6 - 64 bit words memory).

Other Components - Six component fuselage aerodynamic components are calculated from wind-tunnel data sets; forces on the empannages are obtained from lifting line theory, including interference effects. The tail rotor calculation (thrust and torque) is conducted by a modified blade element model as well, since realistic yaw axis representation had often been a weak element prone to critisism by pilots during older simulations.

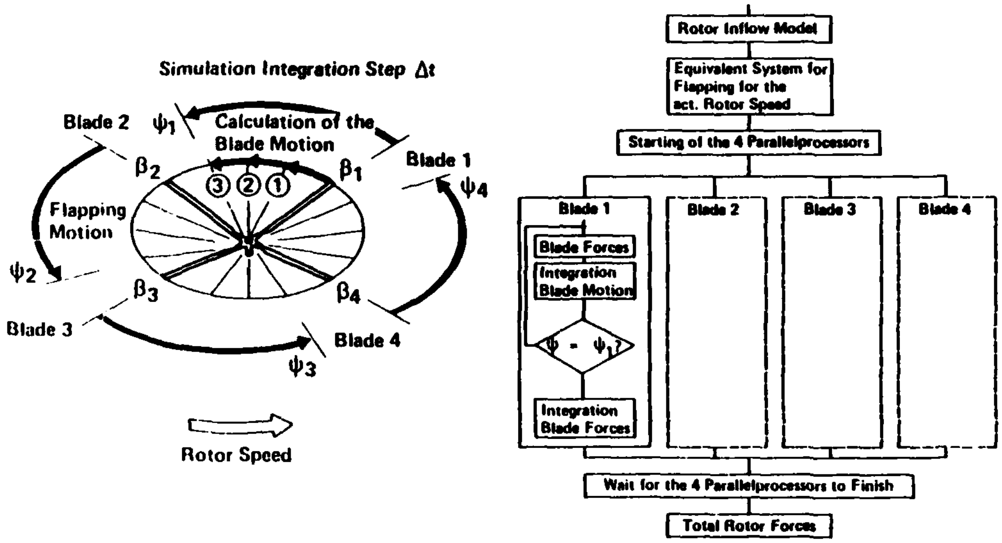

Figure 14 Parallel Calculation of Blade Dynamics (4 Blades)

Power Plant Dynamics - As indicated in Figure 13, power plant dynamics can be well within the range of body and rotor modes, thus influencing the flight dynamics behaviour Reference (5). A simplified engine model is used, composed of first - order engine and governor dynamics with fuel control, specific to each engine type (Figure 15). Piloted simulation results showed that inclusion of rotor rpm/power plant DOF improved simulation substantially.

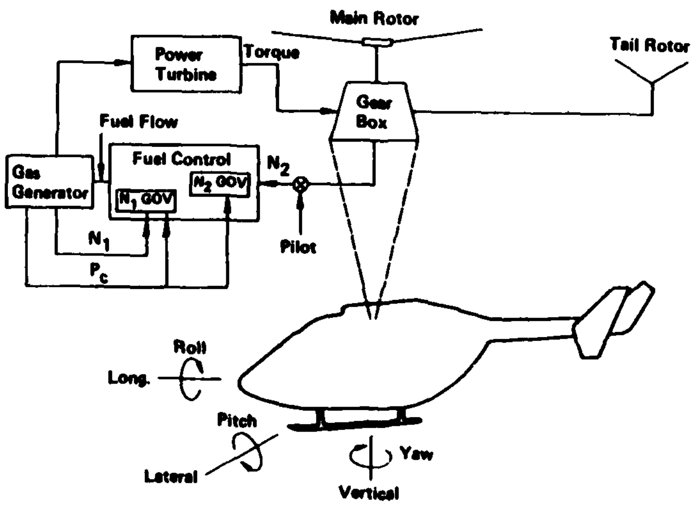

Figure 15 Engine/Fuel Control Model

Atmospheric Representation - To investigate handling qualities and pilot workload the mission tasks have to be simulated under representative turbulence conditions. To quantify possible coupling effects and automatic flight control system performance, defined gusts are required. Both stochastic gust models (Dryden Model) and deterministic gusts are included for specific investigations Reference (6).

A block diagram of the total simulation model is shown in Figure 16.

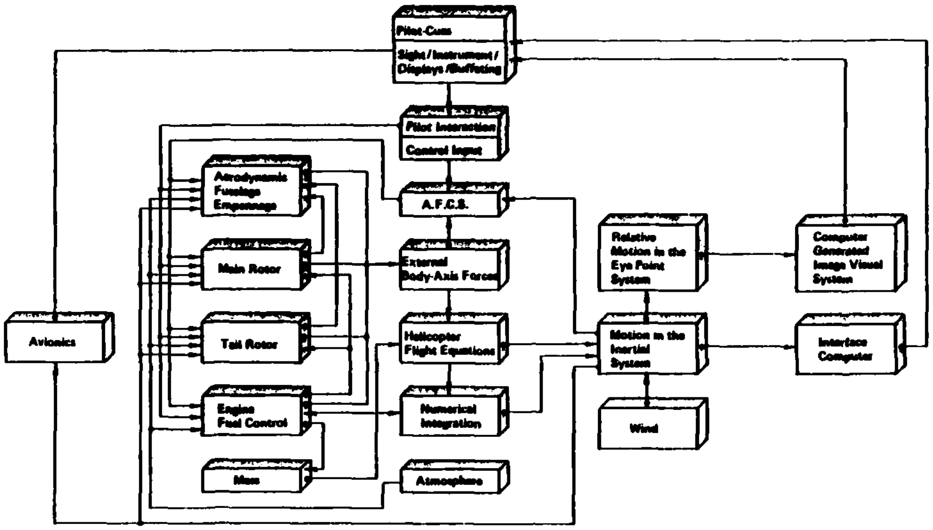

Figure 16 Block Diagram of Total Simulation Model

3.2 VALIDATION OF MATHEMATICAL MODEL

Both qualitative and quantative validation demonstrated high fidelity of the simulations when compared to BO 105 flight test data and pilot's opinions. Using the simulation trim facility steady state values could easily be checked with good agreement. Particular attention was placed on obtaining accurate control correspondence in and around hover a flight state in which a number of complex rotor and interference effects strongly influence handling qualities; a flight state, however, which forms a primary target of the simulation. Comparison of flight test and model cyclic control positions (Figure 17) shows that this was achieved, including the well known rotor downwash complexities contributing to lateral control trim position in low speed forward flight.

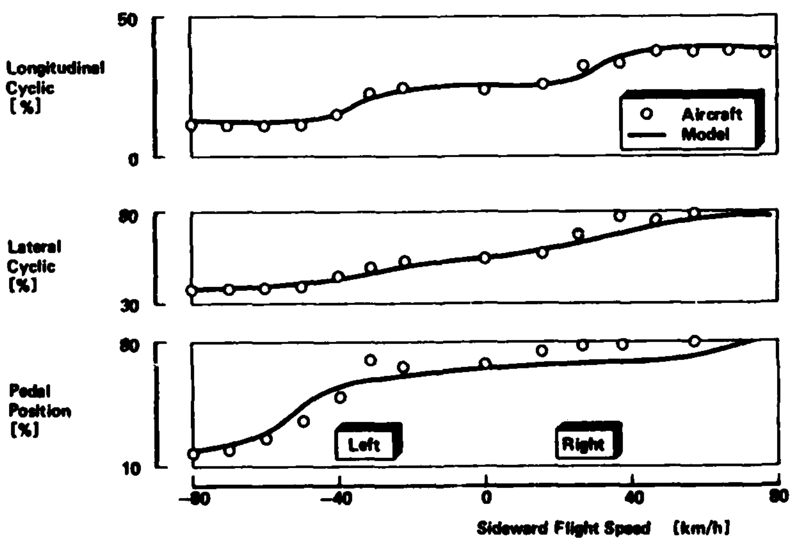

Figure 17 Sideflight Trim Characteristics

Dynamic responses for selected flight cases are shown from horizontal trim for typical speeds at hover, 60 and 100 KIAS in Figures 18 to 19. Comparison of the angular rates and

corresponding attitude responses for the simulation and flight tests again show good agreement. Of equal importance to the quantative comparison are the pilot impressions and acceptance of the simulation. This assessment was achieved by evaluating specific tasks

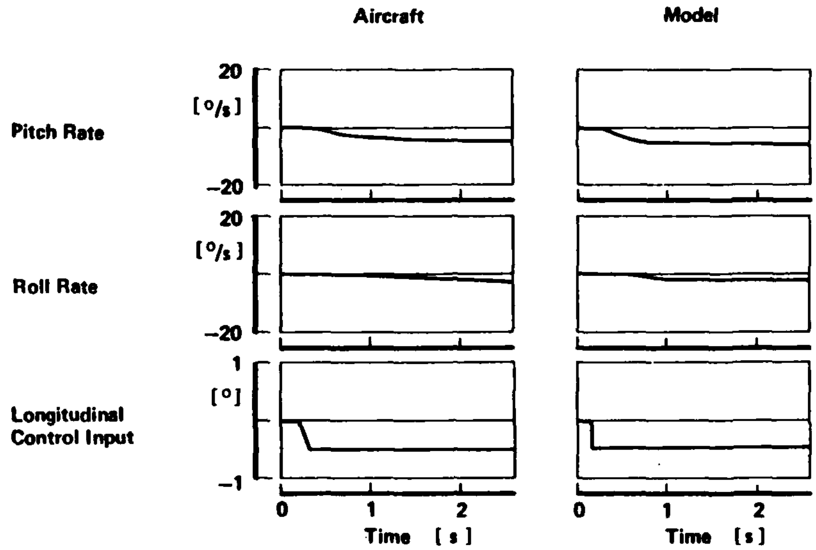

Figure 18 Forward Cyclic Step Input (Level Flight 60 KIAS)

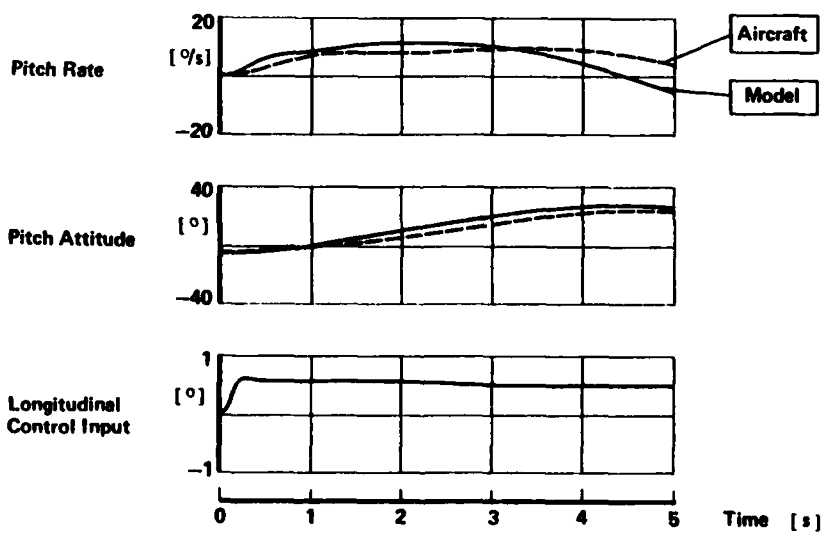

Figure 19 Maneuver Stability (Level Flight 100 KIAS)

and flight phases as tabulated in Figure 20 and which are representative of most missions. Both company and customer pilots were used for this task so that a broad base of opinions could be obtained. In response to pilots comments modelling deficiencies were identified and remedied. Realistic hovering provided some initial difficulties which were traced to the downwash calculation and limits in the lower sideward corner of the CGI causing some difficulty in assessing height above ground. The new CGI dome concept will also produce a much larger field of view. Positive comments were attributed to the accuracy of the control response, trim positions and handling. The visual system was well accepted and harmonized with the attitude and position instrumentation.

- Hover IGE, OGE
- Position Maneuvering
- Slow–Speed Performance (Side, Rear, FWD)
- Normal Take–Off
- Climb to Altitude
- Level Cruise Flight, V_{MAX}, V_{NE}
- Accelerations, Decelerations
- Bank Turns
- Controllarbility
- Dynamic Stability (Long., Lat/Dir, Maneuvering)
- Climb and Descent, AR
- Normal and Steep Approach
- Landing

Figure 20 Gross Maneuvering Pilot Assessment

4. SCOPE OF SIMULATION TASKS AND ACTIVITIES

As discussed in the foregoing sections, particular emphasis during the build-up of the facility was placed on achieving a broad scope of applications. The simulator will be applied during the various design and development phases, under various objectives. A rough summary of the major tasks is given in Figure 21.

- Investigation of tactical flight profiles
- Optimization of 'Teamwork' and minimization of pilot and crew workload
- Optimization of the entire system in the battlefield
  – cockpit design, displays, A.F.C.S
- Configuration Optimization
  – Agility, Manoeuvreability, Handlingqualities, Pilot workload
- Simulation and test of dangerous flight manoeuvres
  – H-V-Diagram, system failures
- Definition of requirements on the Training Simulator

Figure 21 Summary of Simulation Tasks

Mission Optimization - The complexities of modern weapon systems require detailed investigation and critical evaluation at an early stage in the project to ensure effective operation. The simulator provides the tool to realistically represent the battlefield mission and to optimize the helicopter/weapon system with respect to the pilot's cockpit environment. Cockpit procedures related to the weapon system operation, pilot/copilot work-sharing and communication, navigation, interaction with computer terminals are to be investigated. Layout of instruments displays modes and symbology for specific mission phases can be checked. Furthermore, the increasing demand for day/night and all weather capability necessitates the application of electro-optic aids such as LLLTV, IR, night

vision goggles which have to be integrated and compatible with the flight instruments and displays. The concepts of picture and symbology mixing, pilot control and mode selection for the appropriate mission phase are aspects where the simulator can effectively be applied in the helicopter definition.

Configuration Optimization - Aerodynamic and flight mechanics studies during early project stages play an important role in the definition and development phases. Pure digital, unmanned simulations are usually conducted to obtain indications of the main parametric sensitivities of the basic vehicle configuration. The difficulty with such studies is the interpretation of the results and the transfer into actual flight situations. Some of the reasons for this are also related to the fact that current helicopter specifications do not cover many important topics and give only limited guidance for configuration optimization.

The requirements for future helicopters will be based on mission-oriented criteria and therefore, the design optimization of the basic vehicle will require inclusion of the total aircraft system. Currently, configuration optimization studies are being performed for a light transport helicopter project in the following areas: basic configuration design (e.g. rotor characteristics, tail plane geometries, power plant), control system design (e.g. control ratios, harmony (shaping) automatic flight control system), and cockpit optimization (e.g. controllers, displays, external view, avionics).

Simulation during the Development Phase - During the development phase the simulator provides the opportunity for pre-flight test familiarization and training for the pilot. In particular initial flight tests can be "flown" and practiced on the simulator, critical possible failure situations evaluated and emergency procedures defined. During the major part of the development trials flight data can be reevaluated and assessed by the design teams to promote recognition of problems and swift improvements. By following parallel development and continuous "up-dating" of the simulation a basis is provided for defining the necessary training simulator requirements.

System Integration - The complex basic and mission equipments of future military helicopters will require more rigourous capabilities for a total system integration (7). The system integration concept for a current antitank helicopter program is shown in Figure 22. The concept is based on a combined use of integration and the simulation work. Major elements of the integration rig will include the helicopter basic fuselage with original equipments, the integration computer and integration software.

Equipment and subsystems will successively be installed and tested, and systems not at the time available can be simulated by the integration computer. The original cockpit contained in the integration rig is removable and transferable into the CGI-simulation center for inclusion in the piloted real-time simulation.

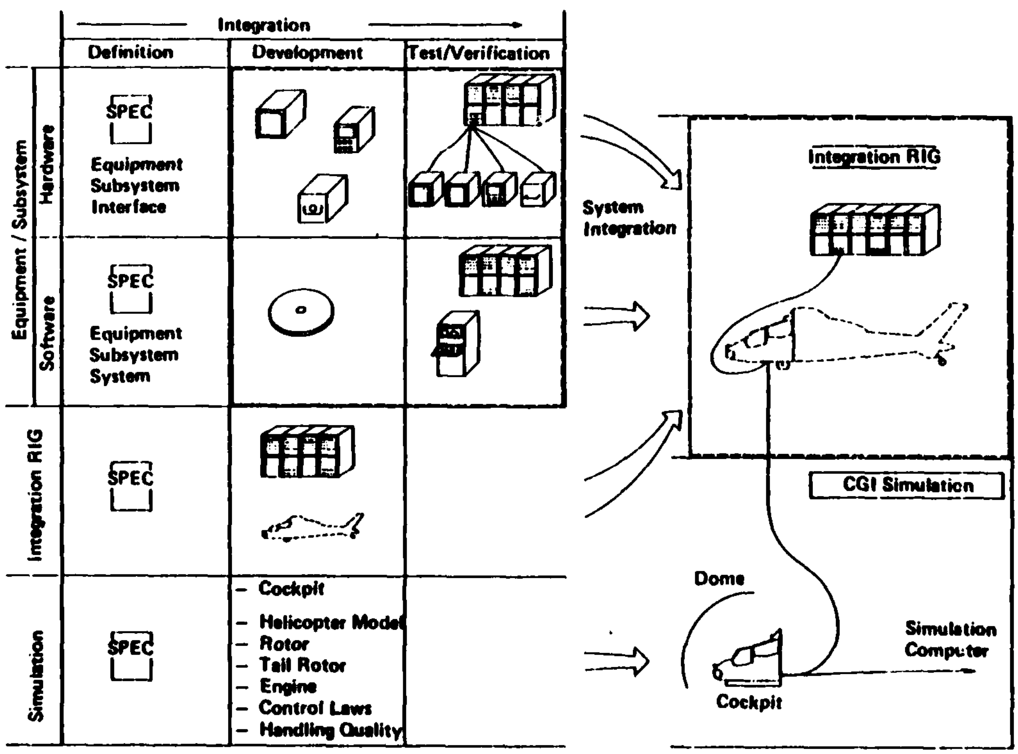

Figure 22 Simulation and Integration Concept

5. CONCLUDING REMARKS

The paper has presented a short description of a new simulation facility for rotorcraft development work. A high degree of flexibility is achieved by an interchangeable cockpit concept, allowing easy change of different vehicles. The computer-generated imagery (CGI) visual system is generally well accepted; some advances in the field of view and projection system will result in further improvements in the near future. The comprehensive helicopter math model represents a complex non-linear technique, which shows high dynamic fidelity and which can approach the flight envelope boundaries. Simulation and integration of hardware on a modular concept provides good basis for evaluation of complex weapon systems.

In general, the current status of the program demonstrates that the simulator facility under development will represent a versatile and highly valuable tool for future rotorcraft development work.

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UN SIMULATEUR D'ETUDES DE SYSTEMES D'ARMES HELICOPTERE

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RESUME

Le Centre d'Electronique de l'Armement réalise au profit de la Direction Technique des Constructions Aéronautiques un simulateur d'études de systèmes d'armes hélicoptère. Ce simulateur restitue les postes chef de bord et pilote et est destiné à l'étude du tir air-air contre hélicoptère à l'aide d'un armement missile et d'un armement canon.

La cabine pilote est fixe et située dans une sphère servant d'écran sur laquelle est projetée à partir d'un tube TITUS une image synthétique de paysage définie par 10 000 facettes (processeur GI 10 000 de SOGITEC). Le champ couvert est de 80° environ et est complété par la projection d'un horizon. La cible est générée synthétiquement par un processeur 250 facettes. La symbologie viseur clair est projetée sur la sphère à partir d'un tube de puissance. Le pilote dispose d'un viseur de casque.

Le poste tireur est déporté et comporte un viseur principal avec présentation d'une image GI 10 000 et les commandes d'armement.

Les sièges sont équipés de vibreurs de siège. L'ambiance de bruit est restituée par un générateur de bruit programmable.

L'ensemble du simulateur est géré par un poste Direction des Essais.

Le Centre d'Electronique de l'Armement (CELAR), dans le cadre des activités de sa division "Evaluation et Simulation de SYstèmes (ESSY)", est chargé d'étudier en relation avec les Services Techniques et les industriels concernés de nombreux systèmes d'armes au profit du ministère de la défense. Ces études sont menées au moyen de la simulation numérique, de la simulation conversationnelle, de la simulation avec éléments réels dans la boucle et enfin des simulateurs. Ces derniers sont développés pour aider à la définition des systèmes en amont de leur conception. Ils doivent de ce fait présenter une grande souplesse d'emploi et être susceptibles d'évoluer pendant leur durée de vie.

Le CELAR réalise ces simulateurs d'études dans des domaines variés et exploite actuellement un simulateur de combat aérien, un simulateur d'interception, un simulateur de tourelle du char futur.

Le simulateur d'études de systèmes d'armes hélicoptère en cours d'intégration permettra d'effectuer des études sur les hélicoptères armés futurs et plus particulièrement leurs systèmes d'armes. Il sera opérationnel en Juin 1984 et permettra de réaliser des expérimentations importantes pour le système d'arme du futur hélicoptère d'appui-protection HAP et à terme de l'hélicoptère anti-char HAC.

Ce simulateur est réalisé au profit de la Direction Technique des Constructions Aéronautiques (Service Technique des Télécommunications et des Equipements Aéronautiques et Service Technique des Programmes Aéronautiques).

En parallèle, le Centre d'Essais en Vol d'Istres développe un simulateur d'étude du pilotage et de l'ergonomie du poste de pilotage.

Les industriels participant à ces travaux sont la SNIAS Division Hélicoptère et le Groupement Industriel des Armements Terrestres.

1 - SPECIFICATIONS GENERALES DU SIMULATEUR

Dans un premier temps, seule la version Hélicoptère d'Appui Protection est à étudier.

L'équipage composé du chef de bord (tireur) et du pilote est disposé en côte-à-côte ou en tamdem dans la cabine.

La fonction principale de l'HAP est le tir contre hélicoptère de jour comme de nuit au moyen d'un armement canon de 30 mm et d'un armement missile Air-Air Très Courte Portée (AATCP). L'AATCP sera la version air-air du missile sol-air très courte portée MISTRAL développé par la société MATRA.

L'HAP pourra être doté d'un armement roquettes non défini actuellement.

L'équipage pour l'acquisition et la poursuite des objectifs pourrait disposer de :

- un viseur principal gyrostabilisé avec symbologie au poste tireur avec poursuite manuelle et automatique,
- un viseur clair avec symbologie au poste pilote,
- des viseurs de casque.

On portera une attention particulière à la restitution de ces matériels et à leurs éventuels différents modes de fonctionnement (image jour, image thermique, image de caméra à intensification de lumière...).

Pour la mise en oeuvre du système l'équipage disposera par ailleurs des commandes de vol et des commandes d'armement.

L'environnement sera restitué de telle sorte qu'il permette le pilotage et l'activation de l'ensemble du système dans des configurations allant du vol tactique au vol à moyenne altitude.

Un poste central de commande appelé le poste de Direction des Essais (DDE) présentera une vue synthétique des combats et permettra de dialoguer avec le simulateur.

2 - OBJECTIFS GENERAUX DU SIMULATEUR

Ils sont au nombre de cinq :

- Organisation du poste de pilotage et du poste tireur pour la fonction tir

  Il s'agit d'étudier la disposition des commandes, la présentation des informations et d'effectuer une mise au point des symbologies de tir.

- Mise en oeuvre du système d'armement

  Il s'agit d'étudier les modes nominaux, les modes dégradés, les procédures d'emploi, les pannes et enfin le dialogue pilote-chef de bord (tireur).

- Evaluation des conduites de tir

  Les trois conduites de tir seront évaluées dans l'ordre suivant : canon, missile, roquette.

- Couplage pilotage-mise en oeuvre armements

  Il couvre essentiellement l'évaluation des modes de pilotage automatique spécifiques.

- Approche des problèmes opérationnels.

Nota : le tir de nuit est à étudier.

3 - DEFINITION DU SIMULATEUR

De l'analyse des spécifications générales, il ressort ;ue :

- la restitution d'un environnement visuel de bonne qualité est indispensable pour tester l'emploi du système en configuration de vol tactique,
- compte-tenu de l'incertitude existant au niveau de l'implantation en cabine de l'équipage, il est possible de déporter le poste tireur ; d'autant que les échanges pilote-tireur s'effectuent essentiellement au moyen des équipements et de la phonie,
- le tireur mettant en oeuvre le système d'arme par l'intermédiaire du viseur principal, il n'est pas indispensable pour ce poste de visualiser le paysage en dehors du viseur.

Le simulateur pour restituer les perceptions visuelle, auditive, tactile se compose des éléments suivants :

- Perception visuelle

  Les formes, les volumes, les aménagements des postes sont représentatifs de ceux de l'HAP.

  Le paysage est créé par un générateur synthétique d'image (GSI) de haute définition (10 000 facettes).

  Les cibles sont élaborées par une machine GSI spécifique pour la présentation au pilote et intégrées dans le paysage pour la présentation au tireur.

  Les images sont :

  . projetées au pilote sur une sphère de 8 m de diamètre par deux tubes de projection : l'un pour le paysage, l'autre pour la cible,

  . présentées au tireur sur un moniteur à balayage TV au travers d'une optique d'adaptation.

- Perception auditive

  Les membres de l'équipage communiquent par la phonie (interphone de bord) et sont également en liaison avec le poste DDE.

  L'ambiance de bruit de l'hélicoptère est restituée par un générateur de bruit programmable.

- Perception tactile

  Les commandes, les poignées armement et les poignées de pilotage sont représentatives de celles de l'HAP.

  L'ambiance vibratoire est créée par un vibreur de siège.

Les évolutions de l'appareil sont ressenties d'une part visuellement et d'autre part par les efforts sur les manches de pilotage à l'aide d'une restitution d'effort hydraulique.

Nota : compte-tenu que la machine GI 10 000 ne sera pas disponible avant mi-85, une solution transitoire est retenue qui exclut le vol tactique. Il s'agit d'un projecteur d'horizon à partir de photographies asservi en roulis, lacet, tangage mais ne créant pas de défilement d'image.

4 - ORGANISATION GENERALE DU SIMULATEUR (cf figure 1)

Sur la figure 1 on distingue :

4.1/ Le centre de calcul temps réel pilotant l'ensemble du simulateur. Le calculateur est un UNIVAC 1180 bi-processeur (1182S), les interfaces temps réel appelées MUTIN ont été développées par SOGITEC. Les principales caractéristiques de ce centre sont les suivantes :

- 2 unités centrales 1100/80. Extension possible jusqu'à 4 mémoires cache : 2 fois 8 Kmots de 36 bits de cycle 200 ns. Mémoire principale 2 fois 524 Kmots de 36 bits, une unité de transition permet de séparer une partie du système,

- interconnexions : elles assurent le contrôle des simulateurs en temps réel en permettant l'échange de données à des débits de l'ordre de 500 000 mots/s sur des distances atteignant 200 m. Le réseau est composé de 6 lignes connectées chacune à un canal. Un coupleur spécifique MUTIN permet de gérer les échanges.

4.2/ Le poste pilote à l'intérieur d'une sphère de 8 m de diamètre servant d'écran.

3 images sont projetées sur la sphère :

- le paysage,
- la cible,
- la symbologie du viseur clair.

4.3/ Le poste tireur avec le viseur principal.

4.4/ Le poste direction des essais piloté, en ce qui concerne les unités graphiques, par un calculateur GOULD SEL 32-27 relié au centre de calcul.

4.5/ La chaîne de génération d'image de paysage comprenant :

- une ou plusieurs bases de données stockées sur des disques 300 Moctets,
- un extracteur d'image,
- un processeur de calcul de l'image,
- un calculateur pilote.

Les points 4.2, 4.3, 4.4 et 4.5 sont détaillés ci-après.

5 - LE POSTE PILOTE

Pour minimiser les coûts de développement du simulateur, le CELAR a proposé pour la restitution du poste pilote une solution proche de celle qui a été étudiée et développée dans le cadre du simulateur de combat aérien.

La cabine du pilote est située à l'intérieur d'une sphère de 8 m de diamètre servant d'écran et sur laquelle sont projetées les images du paysage, de la cible ainsi que du viseur clair.

Enfin pour des raisons de disponibilité du processeur GSI 10 000 facettes (1985) la première étape de développement du simulateur ne permettra pas de restituer le vol tactique.

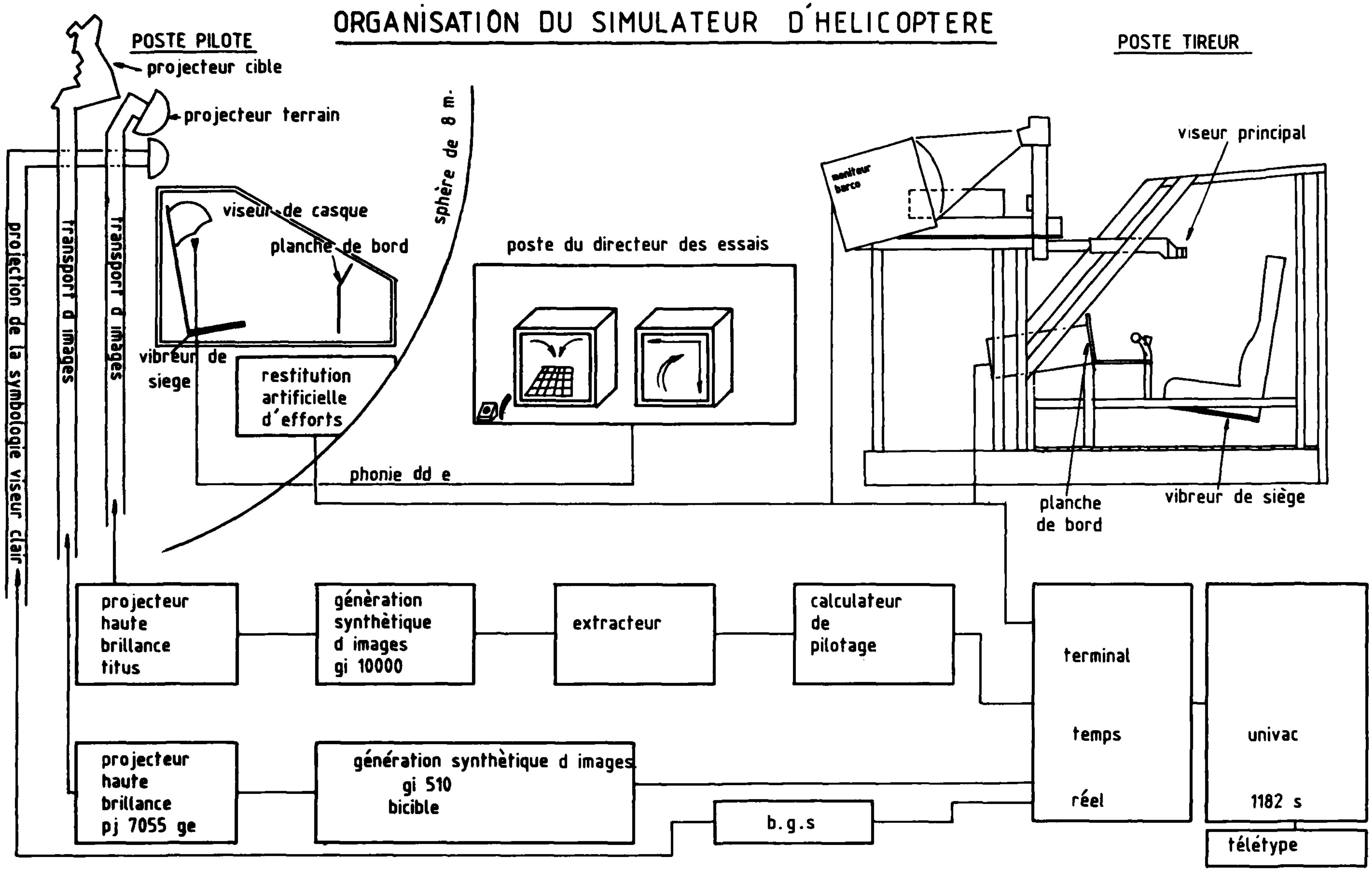
5.1/ La cabine

La forme et l'organisation intérieure de la cabine sont représentatives de celles du poste de pilotage HAP (référence année 1980). La cabine a été réalisée par la SNIAS et compte-tenu de l'état d'avancement du programme HAP à l'époque de la commande des réserves avaient été prises afin qu'elle fût modifiable et reconfigurable.

Les commandes de vol sont prises en compte comme suit :

- simulation des commandes de vol type SA330 puis HAP,
- simulation des systèmes de restitution artificielle d'efforts sur les axes de commandes de vol (électrohydraulique),
- simulation des trims,
- simulation du comportement dynamique et statique des timoneries de transmission et des vérins,
- couplage des commandes et du pilote automatique type SA 330 (première phase) puis HAP,
- conjugaison des commandes de collectif et de lacet,

ORGANISATION DU SIMULATEUR D'HELICOPTERE



- possibilité de modifier rapidement les caractéristiques des commandes (loi effort/déplacement, inertie des commandes, frottement visqueux couplage PA, etc...).

Les trois commandes de vol principales (manches : cyclique, collectif et palonnier) ont été décomposées en trois parties :

- un organe de commande sur lequel le pilote agit,
- un organe de sensation, restituant les efforts ou les déplacements (vérins hydrauliques),
- une unité de calcul élaborant les ordres de l'organe de sensation.

Les appareils de bord restitués sont les suivants : indicateur de radio-sonde, altimètre, anénomètre, variomètre, tachymètre (nombre de tours rotor), accéléromètre, indicateur de pas et limite de puissance, horizon artificiel, indicateur de situation horizontale.

Les équipements complémentaires sont les commandes d'armement situées sur les banquettes latérales, le pilote automatique, les moyens de phonie.

Le siège de type PUMA est équipé d'un vibreur de siège ayant une bande passante de 25 HZ dans une bande utile comprise entre quelques HZ et 50 HZ.

5.2/ La projection cible

La chaîne se compose d'un générateur d'image synthétique, d'un projecteur vidéo de puissance, d'un transport d'image et d'une tête périscopique.

Le générateur d'image synthétique de cible est présenté au paragraphe 8.1.

Le projecteur vidéo de puissance est le PJ7055 de General Electric. Il permet de projeter une image monochrome (noir et blanc) de définition 625 lignes. La puissance lumineuse garantie est de 2000 lumens et le rapport de contraste de 75.

Le transport d'image réalisé par la société CERCO présente une diamètre hors tout de l'ordre de 130 mm et une longueur de 3 m.

La tête périscopique est équipée d'un zoom commandé par trois motoréducteurs d'entraînement : mise au net de 1,4 m à l'infini, ouverture de f/5 à f/22 et focales de 40 à 400 mm (la gamme d'ouverture ou de focales est explorée en une seconde). Le champ maximal sous lequel est vue la cible est de 12°.

La tête est motorisée, les caractéristiques principales des asservissements sont les suivantes :

- débattement 360 degrés,
- vitesse angulaire maximale : 15 rad/s,
- accélération maximale > 200 rad/s^2,
- précision statistique du positionnement du miroir : ± 1,5 milliradian.

5.3/ Le viseur clair

La symbologie du viseur clair est générée par un BGS (cf 6.6). Compte-tenu que la cabine est spacieuse (par rapport à celle d'un avion), que le champ souhaité est important (supérieur à 20°) et qu'enfin les assiettes de l'hélicoptère varient fortement en fonction des conditions de vol ; la solution retenue pour la restitution du viseur clair consiste à projeter la symbologie sur la sphère. Le tube de puissance est un tube BRIMAR 2,5 pouces 34000 cd/m^2.

Cette solution permet en fonction des configurations de vol de caler l'image en jouant sur sa position dans le tube de projection.

5.4/ Projection du paysage

On distingue trois phases de développement :

- de mi 1984 à mi 1985 emploi d'un système de projection de l'horizon donc ne restituant pas le défilement et ne permettant pas d'étudier le combat en vol tactique,

- de mi 1985 à mi 1987 (à préciser) utilisation de la chaîne GSI 10 000 facettes, d'un tube de projection TITUS et d'un champ de projection limité (entre 60° et 80°) complété par la projection de l'horizon,

- à partir de 1987 restitution de l'environnement sur 180° minimum en gisement (solution non arrêtée).

Première phase

Le système comprend :

- un dispositif de projection de la terre,
- un dispositif de projection du ciel,
- une correction des erreurs de parallaxe,
- un système de projection quatre axes.

L'éclairement est réglable de 0 à 5 lux, l'image sol est choisie parmi 6 diapositives placées dans un barillet représentant le sol vu de différentes altitudes (pojection fish eye sur 180°).

Les performances sur l'ensemble des quatre axes sont :

- vitesse maximale > 10 rad/s,
- accélération maximale > 300 rad/s^2,
- précision de positionnement statique : ± 5 minutes d'arc.

Deuxième phase

La chaîne se compose :

- de la GSI 10 000 facettes SOGITEC (cf paragraphe 8.2),
- d'un tube de projection couleur SVS 12 TITUS de la société SODERN de définition 1065 lignes, flux lumineux mesuré 1130 lumens, rapport de contraste mesuré 90, variation locale d'éclairement mesurée 19 %,
- d'un transport d'image REOSC,
- d'une tête de projection motorisée en gisement de champ instantané compris entre 60° et 80°,
- d'un dispositif de projection de l'horizon.

5.5/ Viseur de casque

Il est réalisé par THOMSON-CSF et non spécifique de l'application simulateur. Il se compose :

- d'un casque équipé d'un visuel et d'un jeu d'émetteurs électro-optiques,
- d'un ensemble de deux capteurs électro-optiques avec boîtier auxiliaire,
- d'un boîtier calculateur de direction du champ visuel,
- d'un poste de commande pilote.

5.6/ Phonie, bruit d'hélicoptère

Le pilote dispose d'une boîte de mélange radio comportant un sommateur d'entrée des sons arrivant du DDE, du tireur et un amplificateur d'attaque de ligne pour distribuer le son micro vers le DDE et le tireur.

Le processeur de génération de bruit programmable appelé 4X de SOGITEC comprend :

- une unité arithmétique et logique dotée d'opérateurs rapides,
- une mémoire de formes d'ondes de 64 Kmots de 16 bits,
- une mémoire de données de 1 Kmot de 24 bits,
- une mémoire de microinstructions de 1 Kmot de 52 bits,
- une mémoire d'adresses de 1 Kmot de 10 bits.

6 - LE POSTE TIREUR

6.1/ Structure de la cabine

La cabine maquettée comprend la planche de bord, le pupitre, les banquettes latérales.

Le siège est celui de l'hélicoptère PUMA de la SNIAS équipé d'un vibreur de siège.

Les commandes de vol cyclique et collectif ne sont pas restituées.

6.2/ Planche de bord

On ne s'intéresse pour ce poste qu'à la fonction armement. Le tireur dispose de :

- un écran tête basse à balayage télévision couleur de dimensions 13 cm x 13 cm ainsi que des touches fonctions de commande de l'affichage.
  Les images affichées sont soit des pages de texte permettant de dialoguer avec le système, soit une recopie de l'image vue dans le viseur principal. Le procédé de génération de ces images est explicité aux paragraphes 6.6 et 6.7,

- un panneau de gestion du système d'armes par l'intermédiaire d'un bus et qui comprend des interrupteurs et des voyants d'état et de panne.
  Les commandes figurent également au poste pilote avec uniquement un compte rendu de situation sur des voyants,

- un poste de commande d'armement PCA comprenant quatre parties : mise en oeuvre générale, système canon, système missile, système roquettes. Il s'agit de boutons poussoirs et de voyants.

6.3/ Le pupitre central

Il ne comprend que les boutons de réglage et d'éclairage de la cabine.

Il comprendra à terme le Pupitre Central de Visualisation.

6.4/ Les banquettes

Elles regroupent les fonctions de commande du boîtier téléphone de bord et du boîtier de gestion du viseur principal.

6.5/ Les poignées

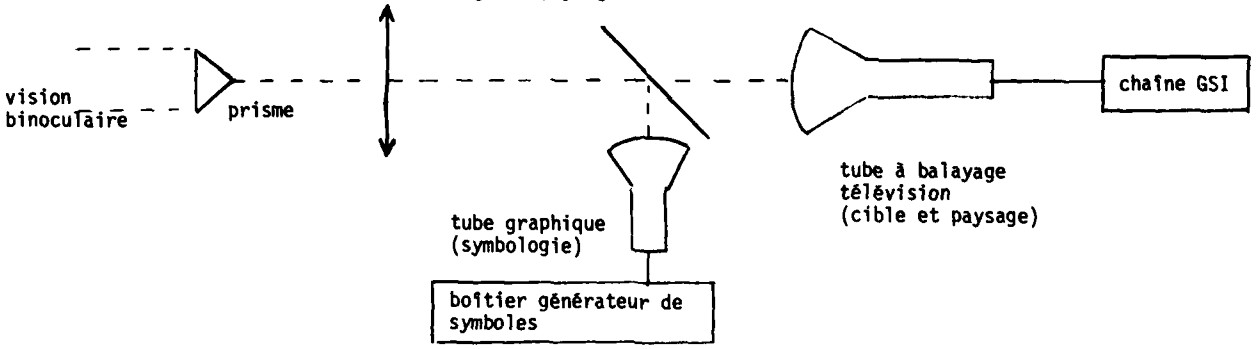
Le tireur dispose de deux poignées armement regroupant les commandes telles que :

- choix du viseur (seul le viseur principal est disponible, le viseur de casque ne pouvant pas être employé),
- choix de l'arme : canon, missile, roquettes,
- modes du viseur principal,
- prise en charge tireur,
- mise en oeuvre du détecteur de veille et d'alerte,
- télémétrie,
- mise à feu.

6.6/ Le viseur principal

Il se compose des éléments suivants :

- l'optique,
- les tubes de visualisation,
- la génération de la symbologie,
- la génération de l'image du paysage et de la cible.



L'optique, réalisée par ANGENIEUX, collimate les images à l'infini, et les mélange.

Ses principales caractéristiques sont les suivantes :

- champ oculaire de 54°,
- FTM 0,5 à 30 cycles/mm dans le plan focal objet des oculaires,
- pas de vignettage,
- luminosité résultante supérieure à 12 % pour chaque oculaire.

Les champ et grossissement sont gérés par le processeur GSI.

Les tubes de visualisation utilisés sont un tube BARCO haute définition (1024 x 1024) et un tube TEKTRONIX.

La symbologie est créée par un système développé par le CELAR et appelé Boîtier Générateur de Symboles (BGS) constitué de :

- une unité de traitement dont l'élément principal est un microprocesseur 68000 MOTOROLA,
- une carte processeur spécifique permettant d'effectuer les opérations géométriques sur les symboles (translation, rotation...),
- des cartes électroniques de la série CONCEPT 60 de la société SINTRA-ALCATEL pour le tracé des vecteurs,
- un logiciel spécifique implanté sur le calculateur UNIVAC de création et d'animation des symboles.

L'image de la cible et du paysage est générée par la chaîne GSI (cf paragraphe 8).

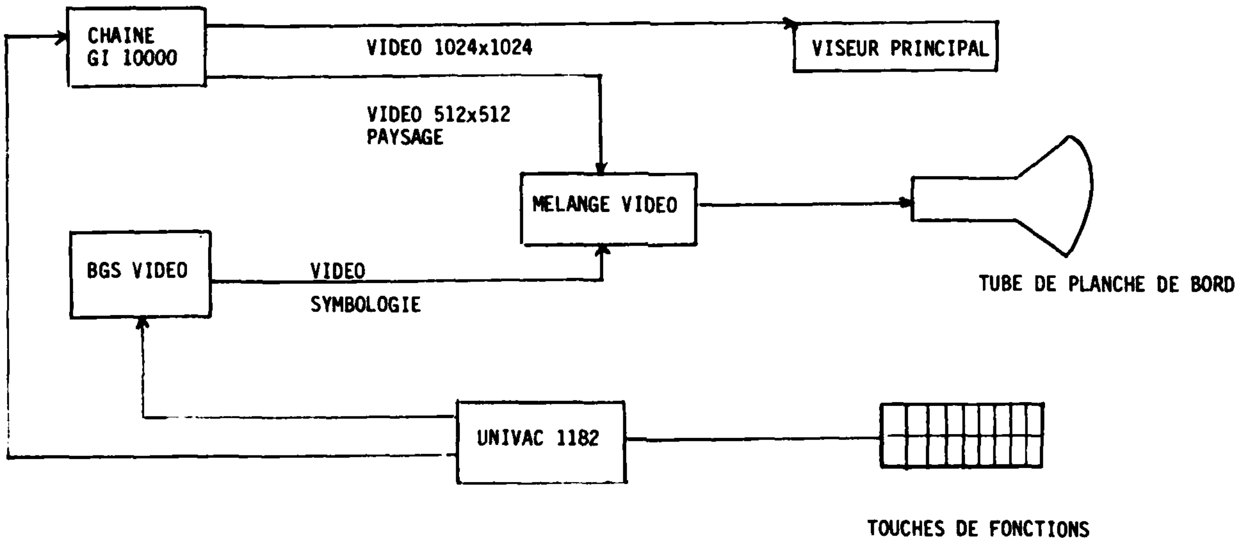
6.7/ Visualisation tête basse

La recopie de l'image du viseur sur l'écran de la planche de bord pose le problème du mixage des deux images précédentes.

Le CELAR a développé une version du BGS adaptée à la commande de moniteurs à balayage télévision dite BGS vidéo. La solution retenue consiste à remplir une mémoire d'image en sortie du BGS de manière à générer un signal vidéo. La version développée utilise deux mémoires d'images 512x512 en flip-flop.

Cette technique ne permet pas actuellement d'obtenir une définition suffisante de la symbologie, c'est pourquoi elle n'a pas été retenue pour le viseur principal.

Le synoptique de commande du tube de planche de bord est ainsi le suivant :



Le mélange vidéo mixe les deux voies soit par un OU classique, soit par un OU exclusif.

6.8/ Génération de bruit et phonie (cf. 5.6)

7 - LE POSTE DIRECTION DES ESSAIS

7.1/ Fonctions du DDE

Les fonctions du poste DDE sont les suivantes :

- initialisation du combat

  choix de la trajectoire de la cible lorsque celle-ci est programmée (plastron). Choix des modes d'emploi de la conduite de tir : modes nominaux, modes secours, modes dégradés. Choix des conditions initiales de l'hélicoptère tireur,

- direction du combat

  au moyen des liaisons phoniques et de consoles alphanumériques reliées au calculateur central, le DDE peut intervenir sur le déroulement du combat :

  . en modifiant les évolutions de l'hélicoptère cible,
  . en imposant des évolutions à l'hélicoptère tireur,
  . en introduisant des pannes,

- présentation d'une vue synthétique du combat

  deux écrans graphiques permettent de présenter :

  sur le premier écran :

  . une vue des trajectoires projetées sur un plan horizontal,
  . ou une vue perspective des trajectoires,
  . ou une vue de la cabine et de la trajectoire de la cible avec pour point d'observation l'oeil du pilote,

  sur le deuxième écran :

  . un domaine de tir,
  . ou un carton,
  . ou une représentation des erreurs de visée.

A ces images sont ajoutées des valeurs alphanumériques nécessaires au Directeur des Essais,

- pilotage de la cible

  un minimanche et un curseur permettent de piloter la cible à partir de vues perspectives présentées sur les consoles de visualisation.
  Le pilotage s'effectue à partir du minimanche ; la commande de vitesse est réalisée par le curseur.
  Cette solution complémentaire au plastron préprogrammé a été retenue de façon à accroître le réalisme du simulateur,

- lancement des dépouillements et rejeu après le combat.

Dans sa version de base le poste DDE est donc servi par deux personnes : le directeur des essais et le pilote.

7.2/ Moyens utilisés

Deux moniteurs à balayage télévision recopiant l'image du viseur principal (paysage et cible + symboles) et le paysage vu par le pilote.

Deux consoles alphanumériques de dialogue.

Un moniteur graphique recopiant la symbologie du viseur principal ou celle du viseur clair.

Des moyens de phonie pour le dialogue avec l'équipage, l'écoute des communications entre les membres de l'équipage, la recopie des bruits de l'hélicoptère.

Un calculateur GOULD SEL 32.27 générant les images du combat à partir des paramètres transmis par le calculateur central UNIVAC. La liaison entre les deux calculateurs est réalisée par fibres optiques.

Deux consoles graphiques VG 2100 SINTRA-ALCATEL.

Un système CONCEPT 60 développé par SINTRA-ALCATEL.

Il s'agit d'un système de visualisation graphique interactif à balayage cavalier et affichage quadrichrome (vert, jaune, orange, rouge) de résolution 1024x1024 points sur l'unité VG2100. La période de rafraîchissement de l'image est de l'ordre de 50 ms.

8 - LES CHAINES DE GENERATION D'IMAGES SYNTHETIQUES

8.1/ La génération d'images de cibles GI 510 de SOGITEC

Ce générateur permet de calculer à la fréquence de 25 HZ deux images cibles indépendantes, définies chacune par 250 facettes planes (triangles et quadrilatères). Les images sont monochromes d'une définition de 512x512 avec une luminance codée sur 8 bits. Cette information de luminance permet d'accéder à une table de fausses couleurs (3 fois 8 bits). La méthode d'affichage des facettes est celle dite "des priorités". Les autres caractéristiques de cette machine sont les suivantes :

- occultation partielle de l'image par positionnement de 0 à 32 rectangles de la couleur du fond, dont les positions et dimensions sont définies par le calculateur de contrôle,
- animation d'un objet composé de plusieurs parties mobiles les unes par rapport aux autres,
- ajout possible d'emports sur les objets,
- représentation de phénomènes lumineux : impact, flash, explosion etc...

Dans le simulateur cette machine permet de créer la cible projetée dans la sphère pilote et de l'occulter partiellement ou totalement selon sa position dans le paysage.

Elle permet aussi de créer la cible présentée dans le viseur principal jusqu'à la mise à disposition de GI 10 000.

8.2/ La chaîne de génération d'images de paysage

La caractéristique essentielle de cette chaîne est de traiter 10 000 facettes en temps réel (une image en 40 ms).

Une facette est un polygone plan de 4 sommets en moyenne et 8 sommets au plus.

La chaîne GSI est constituée des trois éléments suivants :

- la base de données,
- l'extracteur,
- le processeur.

Ces trois points sont décrits ci-après :

- la base de données

elle est créée à partir de trois supports :

. les cartes de l'IGN (Institut Géographique National Français) sur calques orographiques,
. les photos aériennes des zones à numériser,
. une liste ou catalogue des objets typiques.

Les cartes IGN utilisées sont à l'échelle 1/10 000 jusqu'à 1/100 000.

Compte-tenu du fait que l'hélicoptère est susceptible d'évoluer du vol tactique au vol à moyenne altitude, le terrain est numérisé selon trois niveaux de détail dont les densités sont :

- niveau 1 1200 facettes/km^2,
- niveau 2 150 facettes/km^2,
- niveau 3 20 facettes/km^2.

Les paramètres importants d'une facette sont : le nombre de sommets, les coordonnées des sommets, leurs luminances, leurs chrominances, le motif de texture, la nature du terrain ou de l'objet (labour, prairie, béton, sapin, etc...), le niveau de détail auquel elle appartient.

Ces informations sont stockées dans un descripteur de facettes dont la taille est inférieure à 100 Octets.

Lors de la numérisation on définit deux types de facettes : les facettes terrain et les facettes objets. Elles sont assemblées à la création de la base de données et enregistrées sur disque de 300 Moctets.

La base de données ainsi définie est aisément modifiable en base de données ther[illegible]ue. Cette étude a été réalisée par le CELAR et SOGITEC.

La base de données de paysage jour aux normes présentées ci-avant est en cours de réalisation : 100 km^2 sont disponibles, 700 km^2 seront disponibles début 1985.

Extraction de la base de données

Le paysage décomposé en facettes est tridimensionnel. Un engin piloté se déplace à sa guise dans toutes les directions. La zone vue ou observée est à deux dimensions et elle se modifie suivant ces deux dimensions.

La fenêtre d'observation est soit carrée (hélicoptère à moyenne et haute altitude), soit triangulaire (hélicoptère en vol tactique).

Le rôle de l'extracteur est de prélever l'information stockée sur disque magnétique à la cadence de travail du processeur d'image.

Pour limiter l'effet des changements de position des têtes de lecture, on interpose une zone tampon en mémoire vive qui déborde la zone visualisée et dont on rafraîchit la zone de débordement glissante.

Ce matériel a été développé par la société COPERNIQUE à partir de son produit DIRAM 32 et permet d'alimenter le processeur à la période d'échantillonnage de 40 ms avec 15 000 facettes potentiellement visibles pour une fenêtre carrée et 10 000 facettes pour une fenêtre triangulaire.

Les études en cours permettent d'envisager une commutation à l'aide de l'extracteur, base de données visible - base de données infrarouge en moins de une seconde.

Processeur d'image

Les spécifications techniques du processeur sont les suivantes :

- nombre de facettes par image en entrée du processeur	10 000
- nombre de sommets max. par facette en entrée	8
- niveaux de détail	8 max
- optique utilisée pour la projection	linéaire ou non linéaire
- découpage du champ de vue	oui
- lissage des facettes	oui
- texture	oui
- brouillard, brume	oui
- algorithme anti-crénelage	oui
- solution machine à tracer	mémoire d'image
- élimination des faces cachées	par Z buffer
- résolution d'affichage	1024x1024 et 512x 512
- génération d'objets mobiles dans le paysage	4 de 125 facettes max.
- fréquence de récurrence	25 HZ
- temps de réponse	80 ms

Le processeur est en cours de développement et est réalisé par la société SOGITEC. Il sera livré au CELAR début 1985. Les opérations successives à effectuer sont les suivantes :

- prétraitement
  . élimination des facettes à normales arrières c'est-à-dire non visibles du point d'observation,
  . changement de trièdre (base de données vers observateur),
  . délimitation du champ de vue (troncature de l'image),
  . calculs de la projection perspective (passage de l'espace au plan),
  . corrections optiques : il s'agit si l'image est présentée au travers d'un objectif grand angulaire (fish-eye) d'effectuer la transformation inverse avant projection ; ou bien de corriger d'éventuelles aberrations de systèmes optiques,

- traitement
  . générateur de segments
  les facettes planes sont bien adaptées à une représentation sur un écran à balayage télévision. Il suffit de connaître les sommets des arêtes pour en déduire leurs intersections avec chaque ligne de balayage et remplir ensuite de façon automatique les segments des lignes limités par ces intersections,
  . processeur de points
  les segments précédemment définis sont décomposés en 1024x1024 points. Chaque point est défini par un certain nombre de paramètres : luminance, chrominance, Z buffer, adresse de texture et des coefficients de filtrage,
  . mémoire d'image
  elle permet de stocker les caractéristiques des points à afficher. Cette mémoire est gérée en FLIP-FLOP,
  . interface vidéo
  cette interface lit une mémoire d'image pendant que le processeur de points remplit l'autre. Elle génère la vidéo aux standards 1024x1024 et 512x512,

- amélioration des images
  lissage, texture, filtrage. En plus des traitements de base décrits ci-dessus le processeur possède les fonctions suivantes :
  . lissage
  l'éclairement peut varier d'une facette à la suivante et à l'intérieur d'une facette. La fonction d'ombrage calcule la luminance à donner à chaque point affiché,
  . texture
  pour accroître le réalisme de l'image, il est nécessaire de lui plaquer des motifs de texture. Des motifs sont ainsi stockés en mémoire et relus par le processeur de points,

. filtrage

la méthode retenue est la pondération de surface où la couleur d'un pixel est une combinaison linéaire des couleurs qui interviennent dans sa composition.

9 - LE LOGICIEL DU SIMULATEUR

Il est écrit en FORTRAN et réalisé par le CELAR à l'exception du modèle de mécanique du vol développé par la SNIAS.

La période d'échantillonnage de l'ensemble du simulateur est de 40 ms dont 15 ms alloués au modèle de mécanique du vol.

9.1/ Modèle de mécanique du vol

C'est un modèle général de mécanique hélicoptère, l'application à un appareil donné se faisant par introduction des coefficients aérodynamiques et des caractéristiques géométriques et inertielles. Ce modèle permet de représenter l'hélicoptère dans tout le domaine de vol tactique (basse vitesse, grandes incidences, forts dérapages).

9.2/ Modèle de conduite de tir canon

La complexité des algorithmes de conduite de tir conduit à des temps de calcul prohibitifs dans le cadre du simulateur. Le CELAR tente pour surmonter cette difficulté de mettre en oeuvre la méthode du "fichier modèle" ou CSS (Computer Stored Solution) développée par ADERSA-GERBIOS qui consiste à remplacer le modèle par sa réponse pour tout jeu de données d'entrée. L'inconvénient de cette méthode est qu'elle n'est applicable qu'à des modèles figés car toute évolution conduit à rédéfinir pour les mêmes entrées les nouvelles sorties du modèle.

Pour créer le fichier modèle, la méthode suivie se décompose en 3 phases :

- génération d'une base de données représentative d'un sous-ensemble de tous les cas réalisables,
- *génération d'un premier modèle linéaire par morceaux,*
- par passes successives :

  . enrichissement des données dans les zones de risque maximal (là où l'écart entre l'objet à modéliser et le fichier modèle est maximal),

  . redécoupage des zones précédentes,

  . vérification de la validité du modèle pour tout nouveau jeu de données.

9.3/ Autres programmes

Ce sont les divers programmes d'animation des postes pilote et tireur (appareils de bord, poignées, pupitres armement etc...), de l'horizon, des visualisations cibles, paysage, viseur clair, du viseur de casque pilote, du vol missile, d'archivage des informations et de rejeu. Tous ces programmes ainsi que la mécanique du vol et la conduite de tir sont implantés sur UNIVAC. Les programmes d'animation du poste DDE sont implantés sur GOULD SEL 32-27.

10 - CONCLUSIONS

Le simulateur d'études de systèmes d'armes hélicoptère sera opérationnel mi-1984 suffisamment tôt dans le processus de définition de l'hélicoptère HAP pour que les résultats obtenus sur le simulateur puisse être pris en compte dans le programme de développement.

Il permettra, pour la première fois, de faire participer très activement les futurs utilisateurs à la définition du système et bien plus, de tester très rapidement l'impact opérationnel de tel ou tel choix technique.

Enfin, il aura nécessité la réalisation de nouveaux matériels (GSI 10 000 facettes, extracteur) ou l'étude de solutions nouvelles (projection grand champ) qui auront des retombées sur le développement des simulateurs d'entraînement.

ASSESSMENT OF SIMULATION FIDELITY USING MEASUREMENTS OF PILOTING TECHNIQUE IN FLIGHT

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The U.S. Army and the National Aeronautics and Space Administration (NASA) have joined together on a project to conduct a systematic investigation and validation of a ground-based piloted simulation of the Army/Sikorsky UH-60A helicopter. Throughout this investigation, flight testing has been an integral part of the validation effort. Handling qualities and task performance flight testing were performed by the U.S. Army Aviation Engineering Flight Activity (USAAEFA) at Edwards Air Force Base. Systems Technology, Inc., in conjunction with Army and NASA personnel, has used this flight test data base in order to evaluate the fidelity of the UH-60A simulation using the Vertical Motion Simulator (VMS) which is located at the NASA Ames Research Center (ARC). Nap-of-the-earth (NOE) piloting tasks which have been investigated include the bob-up, the hover turn, the dash/quickstop, the sidestep, the dolphin, and the slalom. Results from the simulation indicate that the pilot's NOE task performance in the simulator is noticeably and quantifiably degraded when compared with the task performance results generated in flight test.

This paper is a sequel to the paper entitled "Helicopter Simulation Validation Using Flight Data," presented at the AGARD Flight Mechanics Panel Symposium, "Ground/Flight Test Techniques and Correlation," Cesme, Turkey, October 1982, and published in AGARD Conference Proceedings No. 339 (Ref. 1).

Nomenclature

A	Input height
a	Damping coefficient, rad/sec, in $Y_c = K_c/s(s + a)$
a_z	Normal acceleration (positive up)
c	Control
e	Error
g	Gravitational acceleration, 9.807 m/sec^2
h	Height deviation
i	Input
K_c	Controlled element gain
M	Average absolute amplitude (for each Y_c) of the control response assuming it to be bang-bang with equal positive and negative amplitudes
m	Motion
s	Laplace operator, rad/sec
T_c	Time to complete the control response (i.e., duration of the control response correction for step inputs), sec
T_F	Time duration of final control pulse for $Y_c = K_c/s(s + a)$, sec
T_L	Characteristic time for lead compensation, sec
T_s	Control switching time or time duration of starting control pulse for $Y_c = K_c/s(s + a)$, sec
Y_c	Controlled element transfer function
Y_p	Pilot describing function
Z_w	Heave damping, rad/sec

Z_{δ_c}	Collective control effectiveness, g/percent
δ_{COLL}	Collective control displacement
ζ	Damping ratio
τ_f	aT_F
τ_t	aT_s
τ_s	Transition time delay interval in Fig. 6, sec
ϕ	Phase angle
ϕ_M	Phase margin
ω	Circular frequency, rad/sec
ω_c	Crossover frequency, rad/sec
$\| \|$	Absolute magnitude
$\angle$	Phase angle

Introduction

The results of the flight test and ground-based simulation experiments reported in this paper support a unique rationale for the assessment of simulation fidelity: flight simulation fidelity should be judged quantitatively by measuring pilot's control strategy and technique as induced by the simulator. Measurements of a pilot's strategy and technique are obtained while the pilot is performing specific tasks which demand a given standard of precision with the rotorcraft. The hypothesis is that if the simulator can induce "correct" piloting technique then presumably the simulation fidelity is adequate. Correct piloting technique for the purpose of this investigation is defined as that measured in flight test with a UH-60A Black Hawk helicopter. Hence, this paper offers a quantitative comparison between the piloting technique observed in a flight simulator and that observed in flight test for the same tasks performed by the same pilots.

To begin such a comparison, we adopted and refined a measurement scheme which would be sensitive to pilots' perceptual pathways and piloting technique without interfering with the pilots' actions. The measurement scheme uses sampled data correlation with a sliding "window" in the time domain and least squares estimation to provide time-varying pilot describing functions.

In order to clarify our point of view, we shall first review three definitions.

1. **Validation**--the degree to which transferability is demonstrated between all aspects of simulation performance and corresponding aspects of flight performance.

2. **Verification**--The comparative assessment of the rotorcraft system as it is simulated by its most realistic available model(s). Verification is therefore a necessary but not a sufficient aspect of validation.

3. **Fidelity**--The degree to which those perceivable states are present which are essential to producing correct piloting technique for a given task and environment. Fidelity therefore includes several nearly sufficient aspects of validation.

A researcher quantifies **validation** by providing an answer to the question, "How closely does the simulation reproduce all of those features of flight which are essential to obtaining the desired (and correct) result?" Validation involves all aspects of the rotorcraft simulation, the mathematical model, and the motion, visual, and aural environments. The accuracy of the answer to this question is critical if the researcher expects to use the simulation results to quantify expected benefits prior to actual demonstration in flight.

Verification is quantified through a comparison of the rotorcraft mathematical model(s) and the actual flight vehicle. Many researchers have mislabeled this task as the validation task and assumed that the performance of the mathematical model is the only cornerstone to a successful simulation.

Fidelity is quantified by measuring attributes of piloting technique in both simulation and flight. Poor simulation fidelity may induce negative transfer of training by reinforcing a piloting technique which is inappropriate or incorrect for flight. This occurs through presentation of degraded vehicle, visual, or motion characteristics which induce incorrect pilot psychomotor and cognitive behavior. As an example, the U.S. Air Force has found it desirable to train pilots in air-to-air combat through the use of simulators without motion; this is because the required motion washouts during violent air-to-air combat maneuvers have been shown to induce negative transfer of pilot training to flight.

The results and discussion to be presented in the remainder of this paper will be quantified so as to fulfill the strict requirements for establishing measures of simulation validation, verification, and fidelity. The results, when presented in this format, will be most conducive to drawing conclusions about simulation task fidelity as well as making recommendations for simulator design.

Experimental Design

Black Hawk flight testing was first conducted at the U.S. Army Aviation Engineering Flight Activity (USAAEFA) at Edwards Air Force Base, California, in 1982 in order to evaluate a wide range of mission tasks which were of specific interest for simulation validation purposes. Following a study and an analysis of these tasks, it was decided that the number of tasks on the list should be reduced considerably. Six maneuvers were subsequently chosen for evaluation in the simulation validation experiment. These maneuvers included: the (1) bob-up, (2) hover turn, (3) dash/quickstop, (4) sidestep, (5) dolphin, and (6) slalom.

The bob-up maneuver was evaluated, because it is generally regarded as one of the standard rotorcraft nap-of-the-earth (NOE) scout or anti-tank missile launching maneuvers.

The hover turn task was chosen for evaluation because of its simplicity. Results from the hover turn are still being analyzed; therefore, only pilot ratings will be presented here.

The dash/quickstop, sidestep, dolphin, and slalom maneuvers were chosen for evaluation, because they are all multiaxis NOE maneuvers which are critical to the successful completion of the combat mission. Results from these multiaxis maneuvers are also still being analyzed; therefore, only pilot ratings of these maneuvers will be presented in this paper. The slalom maneuver was not flown in the simulator, because the pilots commented adversely about the computer-generated image (CGI). The visual scene appeared to be constructed in a discontinuous fashion as the pilots maneuvered from object to object. This phenomenon, coupled with the slow visual update rate, was significant enough to make the maneuver unrealistic; therefore, this maneuver was flown and rated in flight only.

The Black Hawk simulation effort was conducted using the Vertical Motion Simulator (VMS) which is located at NASA Ames Research Center in California. The Singer-Link visual system which is installed on the VMS Interchangeable Cab (ICAB) provides a four-window CGI. For the Black Hawk validation study, the data base shown in Fig. 1 was used. Circled numbers on the figure denote the locations from which the various maneuvers were evaluated.

TASK DESCRIPTION

Hover Turns--at point 2 (both 90- and 180-deg hover turns to the center line of the road).

Dash/Quickstop--at points 1 to 2 or points 2 to 3.

Dolphin--from point 3 to point 6.

Sidestep--from point 4 to point 5 and then back to point 4 (both right and left sidesteps were executed).

Bob-up--from a low hover at point 7 to an altitude where the tank could just barely be seen (approximately 16.76 m) and then back down to a hover.

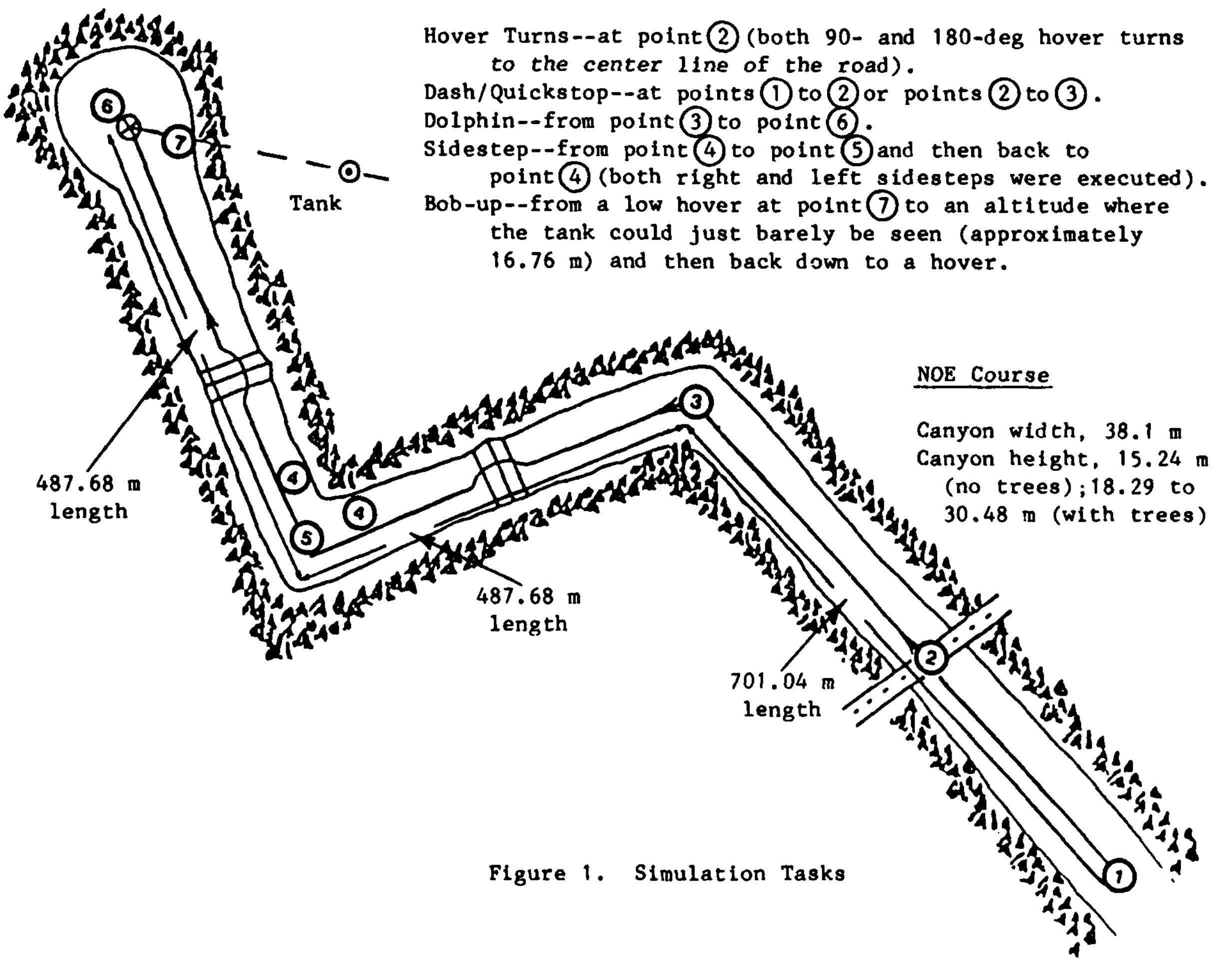


Figure 1. Simulation Tasks

The "standard" simulator configuration included the most up-to-date UH-60A mathematical model available, the four-window CGI, and a motion drive algorithm that provided the

most realistic motion possible. The "standard" motion did not, however, correspond to the motion in flight, as we shall subsequently show, because of the necessary low frequency motion washouts in the simulator. Although it was planned to perform a "full-motion" bob-up in the simulator, time did not permit this test.

Pilot selection was conducted with several requirements in mind. First of all, each pilot had to be familiar with U.S. Army NOE tactics and maneuvers. At least half of the pilots needed to be fully qualified in the Black Hawk, and, most importantly, each pilot had to be available to fly during both the scheduled flight test and the simulation intervals. With these requirements in mind, three USAAEFA pilots (all qualified in the Black Hawk) and two Army pilots (former USAAEFA pilots) on loan to the NASA ARC were chosen. As planned, each pilot subsequently flew both the simulation and the flight test aircraft on numerous occasions for purposes of both pilot training and data acquisition.

Qualitative Pilot Rating Results

Qualitative pilot ratings, as well as pilot comments, were obtained both in flight and during the simulation in the form of Cooper-Harper handling qualities ratings (Ref. 2). A summary of the mean pilot handling qualities ratings (HQRs) for each NOE maneuver (or task) is presented in Fig. 2. These HQRs were obtained in November 1983 at Edwards AFB in winds occasionally gusting up to 40.23 km/h for some of the pilots.

Figure 2 presents a comparison of the mean HQRs and extreme HQR ranges for the flight test and the simulation. All of the tasks were rated at Level Two (3.5 < HQR < 6.5) in the simulator, whereas all were rated at Level One (HQR < 3.5) in flight. When considering these results, one concludes that "something was different" in the simulator which caused a degradation of approximately 1.5 to 2 HQRs. As can be seen, no overlap exists in the rating values. It was of interest to determine the significance of these differences in means while also accounting for the inherent discriminal dispersions in the HQR scale found by McDonnell (Ref. 3). Behrens' test (Ref. 4) revealed that each difference in mean ratings in Fig. 2 was significant at the 90 percent level of confidence.

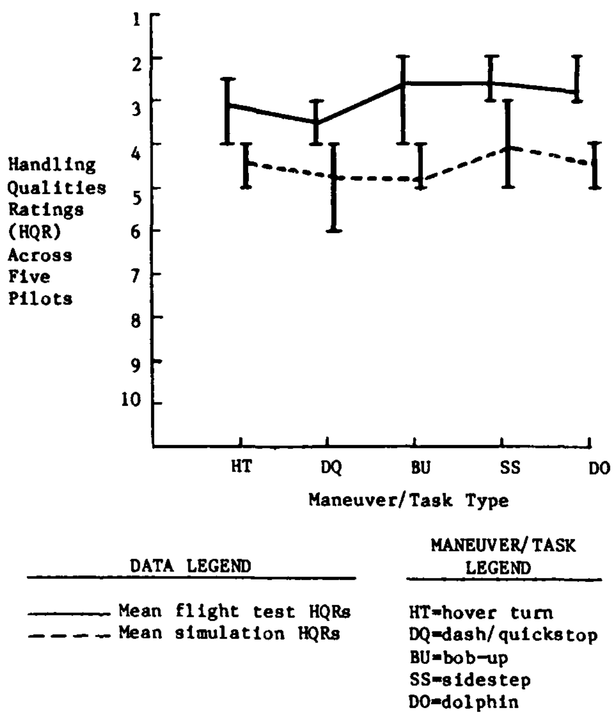


Figure 2. Ranges for Pilot Handling Qualities Rating as a Function of Task for the Flight Test and the Baseline Simulation

A detailed review of the pilot comments identified simulation characteristics which resulted in the higher HQR values being given by the five pilots. The following list presents a summary of the characteristics of the simulation judged by the pilots to be not representative of flight characteristics.

- Larger thresholds of visual perception of movement
- Inability to judge range and height as accurately
- Insufficient scene content and texture in CGI
- Insufficient cockpit field of view
- Deceptive motion cues--only the initial cues feel correct
- Insufficient damping in all axes of simulated rotorcraft
- Vertical PIOs and roll PIOs (PIO = pilot-induced oscillation)
- Exaggerated collective control "pumping"
- Exaggerated cyclic control inputs
- Stabilator-induced dynamics absent
- Excessive pilot workload

Not surprisingly, in reviewing the pilot comments, we can attribute the "problems" to any as well as all of the three major simulation components--the visual CGI system, the vertical motion system, and the Black Hawk mathematical model.

The field of view-, scene content-, and texture-related problems associated with the VMS CGI system (as well as for other CGI systems) have already been identified and discussed in several references (e.g., Refs. 5 through 7). Mention is also made therein of pilots' inability to discern precisely closure rates, rates of change in altitude, and range to objects in the visual scene. The pure transport delay inherent in the CGI is

also a function of the digital computation cycle time. For the Black Hawk simulation, the overall transport delay was 120 milliseconds, which increased the disparity between visual and motion cues. Little could be done to alleviate most of these CGI-related problems, because they are caused by software or hardware limitations. Researchers will simply have to wait for technology to provide (and NASA to procure) more advanced wide angled and externally projected CGI systems which provide more detailed scene content and texture. Fortunately, these types of systems are in development and will be available for use in the future.

The motion-related problems can be viewed from two perspectives: either the motion system is really less capable of generating the desired motions than it should be, or the mathematical model is responsible for generating unsatisfactory commands to drive the motion generation algorithms.

Lastly, the mathematical model may always be suspect when a lack of simulation fidelity exists. The pilot comments clearly identify concern as to the quality of the response damping and control sensitivity. Values for moments of inertia, damping coefficients, the control gearing, and the modeling of the automatic flight control system might also be considered suspect, but one must remember that pilot perception and piloting technique also govern closed-loop responses, so the mathematical model is not the only suspect. Fortunately, for the mathematical model, the verification aspect of quantifying simulation validation can be made in a straightforward manner.

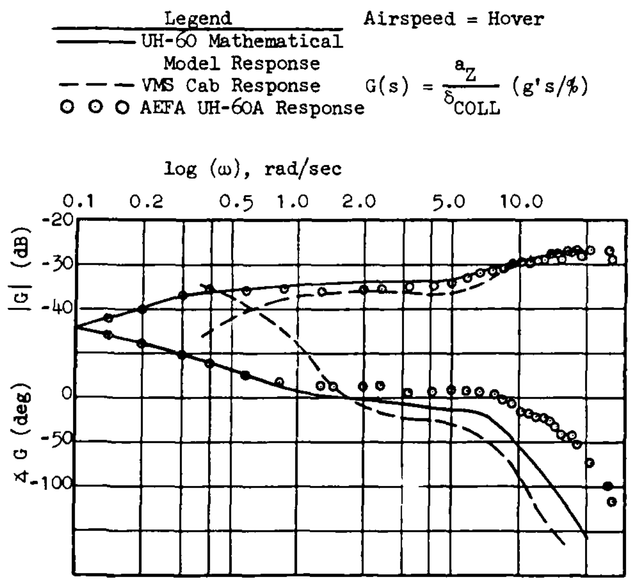


Figure 3. UH-60A Rotorcraft and Simulator Frequency Responses for the Transfer Function a_Z/δ_{COLL} at Hover

Verification

Prior to the formal simulation experiment, we conducted a special fixed-base off-line simulation for the purpose of verifying parts of the mathematical model with the pilots involved. Black Hawk pilots from USAAEFA helped in this phase of verification. Trimmed, hovering, and level flight simulation results compared quite well with flight data taken by USAAEFA pilots in Ref. 8. The pilots were also helpful in matching the "feel" of the VMS cab controls with the feel of those controls in the Black Hawk (S/N 77-22716) with which they were intimately familiar and which was to be used in the flight portion of the experiment.

An example of the dynamic response correlation among the mathematical model, the VMS cab, and the flight data is presented in Fig. 3 in the form of logarithmic frequency responses for vertical acceleration with collective control excitation. Mathematical model and flight responses in the low frequency decade (up to 1.0 rad/sec) are based on transient responses, and above 1.0 rad/sec, on frequency sweeps using the collective controls. The VMS cab response is based entirely on frequency sweeps.

The bob-up was identified by the pilots as one of the maneuvers in the simulator which was least like the corresponding flight maneuver. Of particular interest to us was the comment that heave damping seemed to be significantly less in the simulator in comparison with flight. The same comment and HQRs resulted from the simulated bob-ups reported in Ref. 13 with nearly the same motion washout. An analysis of the Black Hawk mathematical model response characteristics and of flight test step response data verified that heave damping ($-Z_w$) was approximately equal to 0.3 rad/sec and that collective control effectiveness ($-Z_{\delta_c}$) was very close to 0.02 g/percent for both the mathematical model and the rotorcraft.

The pilots' unit gain crossover frequencies for height regulati ı are between 0.5 and 1.5 rad/sec. The mathematical model and the flight responses are in good agreement over this range. The second-order vertical motion washout at 0.3 rad/sec, however, provides a significant amount of phase advance in the cab motion over the whole range of identified crossover frequencies. At a frequency of 0.6 rad/sec, motion phase advance in comparison to the mathematical model is approximately 80 deg. At 1.0 rad/sec, the motion phase advance decreases to approximately 50 deg. Cab motion is in phase with the mathematical model only at approximately 1.65 rad/sec, above which cab motion lags the mathematical model. From this motion angle advance must be subtracted (algebraically) the additional phase lag caused by the 120 millisecond transport delay between the mathematical model and the visual scene, which is -7 deg at a frequency of 1.0 rad/sec and otherwise proportional to frequency. Thus the disparity between the phasing of the visual cues and motion cues is increased even more and may have an impact on pilot cueing to such an extent that it could be responsible for some of the reported pilot rating discrepancies.

Quantification of Pilot Strategy and Control Technique in the Bob-Up Maneuver

Convinced that the perceived controlled element may have been different between the simulator and flight, we initiated an analysis of piloting technique.

Instructions given to the pilots for performing the rapid response phase of the bob-ups and hover turns were intended to instill a sense of urgency approaching a step function for the internal height command while the pilot was otherwise engaged in steady-state regulation of his attitude, heading, height, and position in the presence of atmospheric turbulence. It has been found (Ref. 9) that such combined transient and steady-state situations can be represented adequately with a single input dual-path structure such as that shown in the simplified diagram of Fig. 4. To represent a multi-loop situation, the signals shown in this block diagram could be considered as vector quantities. The quasi-linear steady-state path is the one used for regulating errors caused by random inputs or disturbances. It operates when the error (e) has been reduced within a tolerance acceptable to the pilot for the task of regulation. The feedforward element operates on the large transient errors induced by the pilot's internal command or desire to initiate the bob-up.

The roles of the switching and the feedforward element are, in the simplest terms, such as to partition the pilot's control strategy into three phases, each having a different system organization. As an elementary example, consider the typical system step response shown in Fig 5. In terms of the three phases, the operation of the dual path model can be expressed in the following terms:

1. Transition from quasi-linear path to feedforward path, corresponding to the time delay phase of duration τ_t.

2. Patterned feedforward response, corresponding to the rapid response phase of duration T_c.

3. Return to the quasi-linear path, corresponding to the error reduction phase of indefinite duration.

The time delay phase is observable only when the transient forcing function is imposed on the pilot unexpectedly from an external source. It was therefore not observable in the tests reported here.

For the bob-up maneuver, $i = h_i$, $e = h_e$, $c = \delta_{COLL}$, $m = h$, and the quasi-linear steady-state model is Y_{P_h}.

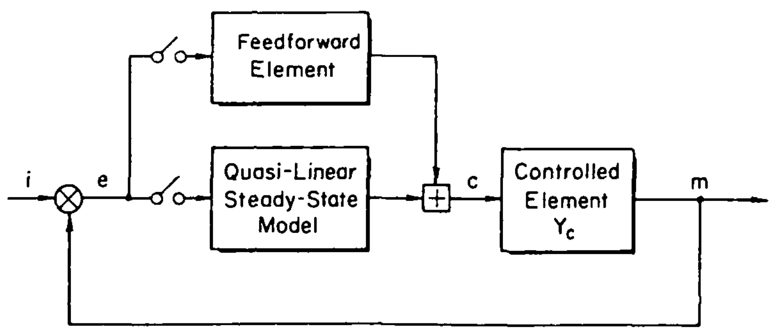


Figure 4. Structure of the Dual-Path Model

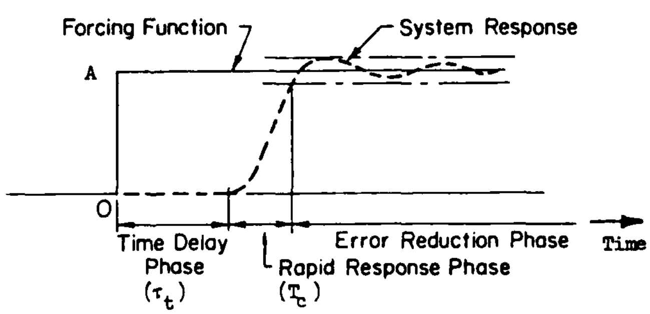


Figure 5. Typical System Step Response

The Bob-Up Rapid Response Phase

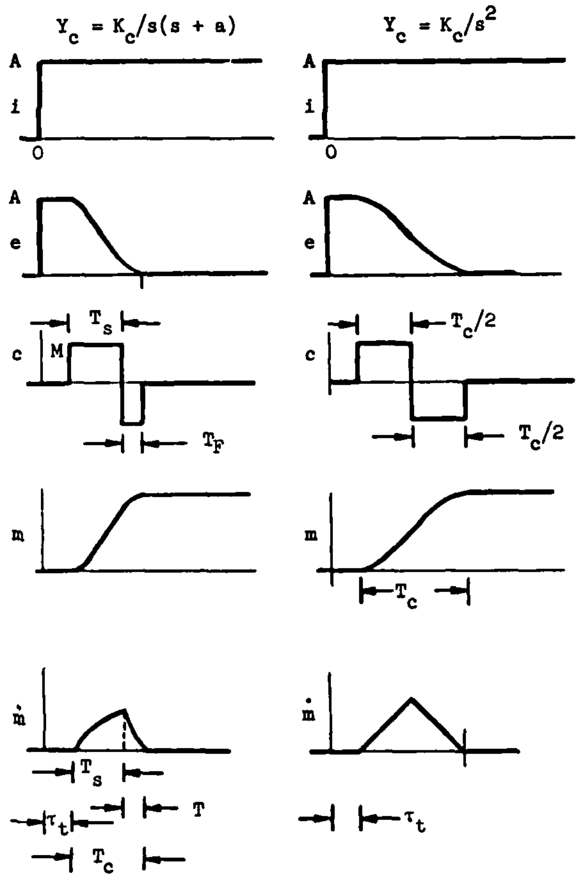
The bob-up rapid response phase begins at the point in time that the pilot mentally defines the magnitude of the error in altitude required to reach a desired bob-up height and decides to raise the collective control to begin the bob-up. This is essentially an open-loop command which is designed by the pilot to obtain a rate-of-climb as soon as possible while keeping within safe torque limitations and within the capability of the rotor speed governor. Initially, no attention is given to the problem of stabilizing at the new altitude.

The most important aspect of the rapid response phase is the pulse-like "bang-bang" nature of the control movements. In fact, the pilot's control displacement (c) is a remarkably good approximation to the controller properties of the single-input, single-output, time-optimal control system with $|c(t)| < M$. Here, the bound (M) may represent either a physical limit on the control deflection or, more likely in the piloted case, an implicit restraint imposed by the pilot for the given situation.

The output of the feedforward element for a skilled pilot is peculiar to each controlled element form. For the helicopter, the controlled element transfer function representing height response to collective control displacement is given with good approximation for our purposes by $Y_c = K_c/s(s + a)$. Ideal time-optimal traces for comparison with the actual piloted responses are shown in Figure 6. (Those for $Y_c = K_c/s^2$ are from Ref. 9.) Suboptimal control techniques are evidenced by irregular trapezoidal pulses of long duration and unequal amplitude. Usually the starting pulse is greater in amplitude than the final.

Our first interpretation of bob-up data focuses on the distribution of the control pulse intervals. Note that the optimal intervals are unequal for controlled element $K_c/s(s + a)$; therefore, we would expect the distribution of intervals from an ensemble of bob-ups in flight to be bimodal, since $K_c/s(s + a)$ represents the predominant quasi-linear height response dynamics to collective control, where $a = -Z_w$, the heave damping and $K_c = -Z_{\delta c}$, the collective control effectiveness. The results in Fig. 7 show that the distribution of control pulse intervals is indeed bimodal in flight and corresponds approximately to the theoretical optimum with one exception.

Figure 8, however, shows that the distribution of control pulse intervals for the bob-ups in the vertical motion simulator is unimodal, which is more appropriate for controlled element K_c/s^2. In fact the mode of the observed distribution is approximately equal to the optimum ratio 0.5 on the ordinate in Fig. 8. One interpretation is that the pilots do not perceive heave damping in the cockpit of the vertical motion simulator, even though the mathematical model of the controlled element for height response to collective has been identified as $0.6/s(s + 0.3)$ in both simulator and flight. This interpretation is at least consistent with numerous pilot complaints about a "severe lack of damping about all axes." The mode of the distribution of control pulse intervals is shifted slightly to the left of the line of perfect equality between observation and theory, because the theoretical optimum control pulse intervals ($T_c/2$), calculated from the observed values of A, K_c, and M, tend to be slightly less than the corresponding observed control pulse intervals ($T_s + T_F$) for each bob-up in the simulator.



i = input = A u(t) e = error
c = control m = motion

Figure 6. Ideal Time-Optimal Response Characteristics (T_c = time to complete control program)

Bob-Up Error Reduction Phase in Flight and Simulator

The bob-up error reduction phase, as shown in Fig. 5, begins as the pilot approaches his desired bob-up altitude and switching is made in the pilot model to the quasi-linear steady-state path in Fig. 4. The controlled element (Y_c) for the error reduction phase remains $K_c/s(s + a)$, which requires only first-order lead-lag equalization by the pilot for closed-loop regulation of height in the presence of disturbances. For a combat task (e.g., target designation), a pilot desires to minimize his exposure above his concealing terrain features; therefore, minimization of overshoots in reaching the desired altitude is critical.

Among the nine out of thirteen bob-ups in flight which were considered aggressive, nearly all had an initial overshoot of 1.5 m or less, and two-thirds had an initial overshoot of 0.9 m or less. After a stabilized height was obtained in flight, the height did not vary by more than plus or minus 0.6 m.

The error reduction phase of the simulated bob-ups exhibited more aggressive compensatory control techniques, lower damping ratios, and larger dynamic errors than in flight. The initial overshoots averaged about 3 m. Six of the fourteen simulated bob-ups had initial overshoots between 3.7 and 4.9 m. The simulated error reduction phase exhibited closed-loop damping ratios approximately half those in flight as shown by the regions of complex eigenvalues in Fig. 9. This fact appears to be the basis for the pilots' comments about lack of heave damping, but we believe that the underlying reason is revealed by identification of the piloting characteristics in Table 1.

The consistent identification of lead compensation and crossover frequency provided valuable information in an effort to quantify why pilot performance was significantly worse in the simulator than it was in flight. Many researchers have questioned whether the 120 msec delay in the CGI visual is the culprit for poorer pilot performance on the simulator. Looking at the results of the heave axis bob-up task in this experiment, however, one would have to say "not by itself." The highest identified pilot crossover frequency was approximately 1.3 rad/sec, and this translates into an additional visual phase lag of only -9 deg with respect to the mathematical model. Although this additional visual phase lag increases the disparity between the motion cueing and the visual cueing, this is not enough lost phase margin to be the primary cause for the degraded pilot performance seen.

The mathematical model, as discussed in detail earlier, also does not, by itself, seem to be a primary cause for degraded pilot performance. Nevertheless, the fact that lead compensation is required by the task may contribute to the degraded pilot opinion and degraded performance if that lead compensation is being generated predominantly in the visual modality, because the pilot's effective transport delay increases from approximately 0.3 sec to 0.5 sec when compared to a technique involving complementary lead compensation via the motion and visual modalities (Ref. 10). The different ranges of effective time delay in Table 1 are consistent with this phenomenon. Until the pilot overshoots the height of the canyon rim, there is insufficient scene content from which to generate lead compensation in the visual modality. (The rotor tip path plane, which helps the pilot to prevent overshoots, was not provided by the CGI.) This fact may account for the larger overshoots in the simulator. Thus the pilots' ratings, comments, characteristic lead compensation frequency, larger effective time delay, and lower closed-loop damping ratio in the simulator represent consistent evidence for their failure to employ motion feedback in the simulated bob-up in a way which complements visual feedback.

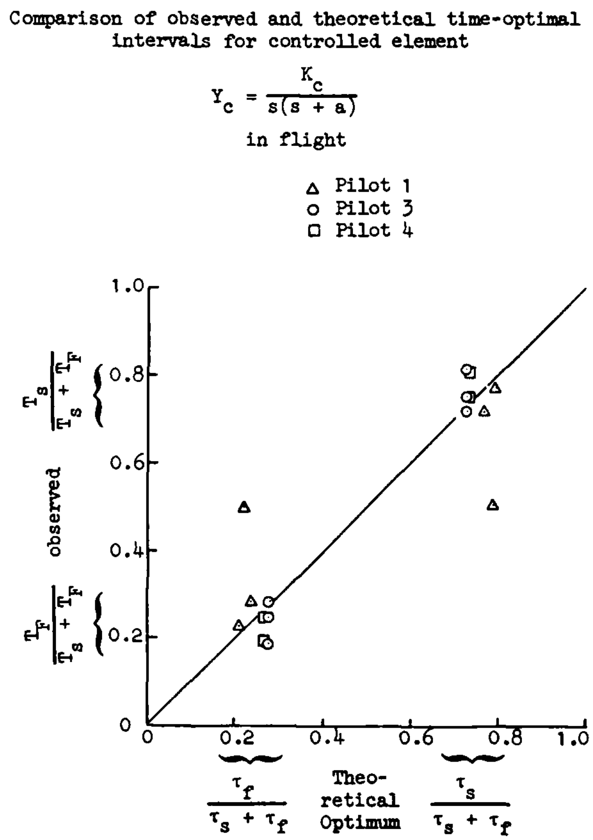


Figure 7. Control Pulse Intervals During the Rapid Response Interval in Bob-Up Maneuvers in Flight

If the generation of the necessary and sufficient first-order lead compensation is based on motion cues, effective integration of the acceleration cues is customarily required in flight, but, in the simulator, the dynamics of the second-order motion washout algorithm must be taken into account in our interpretation (Refs. 11 and 12). Over the portion of the crossover frequency range from 0.5 to 1.0 rad/sec, the phase advance generated in the acceleration cues by the washout is 82 to 50 deg, respectively, with respect to the mathematical model in Fig. 3, and 85 to 57 deg, respectively, with respect to the visual CGI. This magnitude of motion phase advance in the crossover range certainly has the potential to confuse the pilot. Thus the adopted control technique in the simulator may perforce effectively ignore motion cues; such a conclusion is at least compatible with the evidence.

Further investigation of the influence of motion cueing yielded several other interesting results. The bob-up evaluation for a small-motion case yielded an average HQR equal to that of the standard-motion case. A very limited evaluation of a medium-motion case by two of the five pilots yielded an average HQR of 3.75, which is lower (better) than the value for the standard- and small-motion cases. A subsequent qualitative evaluation of this observation was made recently by STI on a Navy helicopter shipboard landing experiment using the VMS (Ref. 6). Pilot comments during that simulation also tended to indicate a preference for a medium-motion configuration over the standard- and small-motion configurations. Collectively, these results suggest that a vertical axis gain reduction in the motion system and a corresponding improvement to the observed phase distortion may be more acceptable for rotorcraft simulations than the present VMS standard-motion configuration. A case can also be made for the evaluation of some form of a nonlinear motion washout algorithm based on the value of the height command being supplied to the motion system by the mathematical model.

It is also important to note that all of the bob-down maneuvers in flight, which were usually slightly less aggressive than the bob-ups, demonstrated highly damped closed-loop dynamics. The slightly reduced aggressiveness resulted in no detectible overshoots for any of the bob-downs.

Bob-downs in the simulator were not analyzed for primarily one reason; that is, an agreement by all five pilots that the lack of texture or scene content for height perception on the CGI made the task unrealistic. While the bob-up task could be performed quite realistically from behind a canyon wall to acquire a tank target over a ridge, the bob-down task was performed behind the ridge, resulting in all four windows of the cab being filled with a monotone green color. The only significant detail in the visual scene was created by the intersection of the different edges or planes of the canyon floor, the canyon wall, and the top of the canyon wall with a monotone blue sky. Without the aid of a radar altimeter, geometric altitude with respect to the terrain was almost impossible to perceive during the bob-down; consequently, the pilot performance was not very good. It was therefore decided that nothing meaningful would be obtained through analysis of the pilot describing function for the bob-downs. One valuable lesson learned from this is that it is very important to provide in the CGI the scene content necessary for performing each NOE flight task. In the present instance, however, the CGI, with a delay of 120 milliseconds, was not even capable of providing the additional scene content required for the bob-down.

Bob-Up Roll PIO Observations

The observation was also made during evaluation of the simulator bob-ups that, for approximately half of the runs, a pilot induced oscillation (PIO) appeared in the roll axis following initiation of the bob-up. This PIO was generally not damped out during the error reduction phase. In reviewing the flight data, no PIOs in roll were found. An analysis of the pilot describing functions in the simulator showed that first-order lead compensation was again generated by the pilots at or above their roll crossover frequencies, and the average crossover frequency for five runs was approximately 1.4 rad/sec. At this crossover frequency, the roll motion-to-visual phase disparity is 60 deg, no phase margin exists in the visual modality, and the pilot's gain has been set so high that roll attitude regulation is unstable. Again the catalyst may be the disparity between motion cues and visual cues caused by the second-order roll rate washout (at 1.0 rad/sec) and the visual CGI delay. Therefore, when the roll axis is excited either by the bob-up maneuver or by turbulence, the only way for the pilot to arrest the PIO in the simulator is to reduce his gain and let the PIO subside.

Summary of Results for the Bob-Up

In summary, the analysis of the bob-up maneuver yielded the results listed below.

1. All pilots rated the flight bob-up 1.5 to 2 HQR points better than in the simulator, and the precision of the maneuver was significantly better in flight than in the simulator.

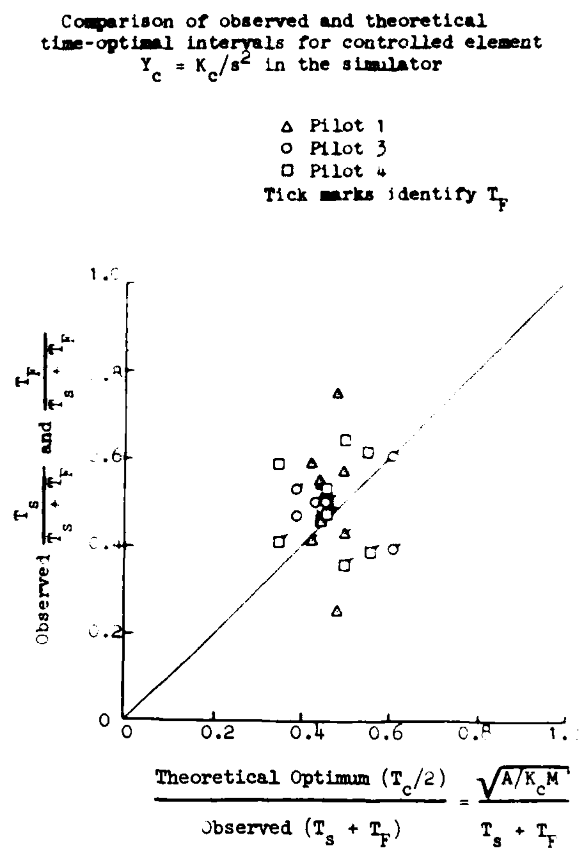


Figure 8. Control Pulse Intervals During the Rapid Response Interval in Bob-Up Maneuvers in the Simulator

Results From the Rapid Response Phase

2. In flight (with one exception), pilots' pulse-like control techniques were appropriate for the actual damped acceleration controlled element transfer function, $K_c/s(s + a)$.

3. In the simulator, pilots' pulse-like control techniques were more appropriate for the undamped acceleration controlled element transfer function, K_c/s^2, even though the mathematical model represented $K_c/s(s + a)$.

4. The results confirm the pilots' comments about lack of heave damping in the simulator.

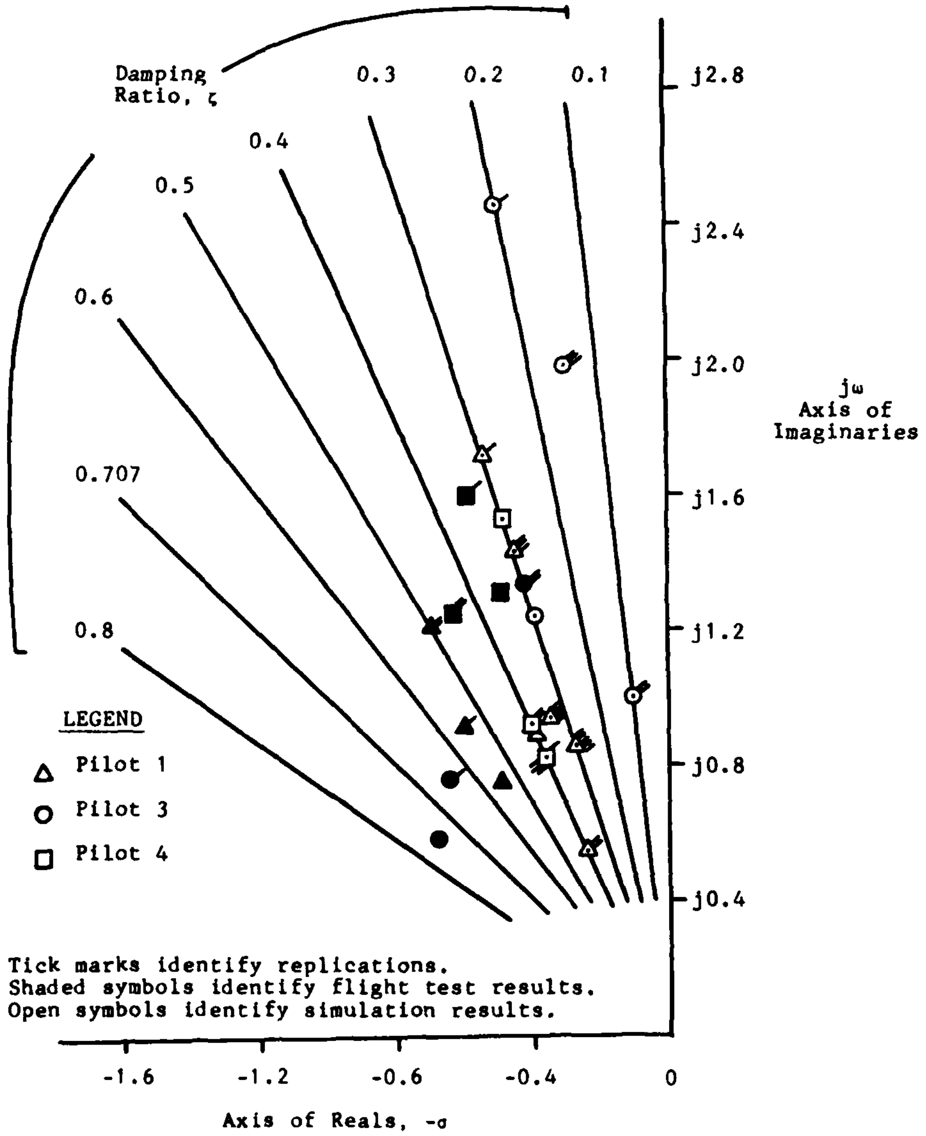


Figure 9. Complex Plane Showing Observed Closed-Loop Characteristics of Bob-Ups During the Error Reduction Phase

TABLE 1. OBSERVED PILOTING CHARACTERISTICS IN BOB-UPS DURING THE ERROR REDUCTION PHASE

Characteristics	In Flight	In Simulator
Gain Crossover Frequency	0.7 to 1.1 rad/sec	0.5 to 1.3 rad/sec
First-Order Lead Equalization Frequency	0.3 to 1 rad/sec	0.2 to 0.4 rad/sec
Effective Time Delay	0.2 to 0.3 sec	0.44 to 0.6 sec
Closed Loop Damping Ratio	0.3 to 0.8	0.1 to 0.4
Roll PIOs	None	≈ 1.4 rad/sec in half of the bob-ups
Overshoots	< 1.5 m	3 to 6 m

Results From the Error Reduction Phase

5. During the error reduction phase in flight, the pilots adopted unit gain crossover frequencies between 0.7 and 1.1 rad/sec; whereas, in the simulator, their adopted crossover frequencies tended to be more variable among runs, between 0.5 and 1.3 rad/sec. Consequently, the ranges of closed-loop natural frequencies tended to be more restricted in flight (0.9 to 1.8 rad/sec) than in the simulator (0.6 to 2.5 rad/sec).

6. During the error reduction phase, all pilots in the simulator developed first-order low-frequency lead equalization in the vicinity of heave damping ($\approx$ 0.3 rad/sec); whereas, first-order lead equalization by the pilots identified in flight was between 0.3 and 1.0 rad/sec.

7. During the error reduction phase in the simulator, the pilots' effective time delay ranged from 0.44 to 0.6 sec, roughly twice that during the error reduction phase in flight. This implies predominant use of the visual modality in generating lead compensation in the simulator.

8. Consequently, average closed-loop damping ratios tended to be higher in flight (0.3 to 0.8) than in the simulator (0.1 to 0.4) because of the greater effective time delays in the simulator associated with both the pilots' low-frequency lead equalization and the visual computer-generated imagery.

9. Roll PIOs appeared in half of the simulated bob-ups but never in flight.

10. Lack of visual height and sink rate cues in the simulated canyon made bob-downs totally unacceptable in the simulator; whereas, in flight, the bob-down was just as precise as the bob-up.

11. Visual height and ascent rate cues were also degraded during bob-ups in the simulator until the pilots overshot the height of the canyon rim by 3 to 6 m. The rotor tip path plane, which helps the pilot to prevent overshoots, was not provided by the visual simulator. There were few overshoots of the hangar height in flight--at most, by only 1.5 m.

Conclusions From the Pilot Ratings of Five Nap-of-the-Earth Maneuvers In Flight and In the Vertical Motion Simulator

All of the tasks were rated at Level Two (3.5 < HQR < 6.5) in the simulator; whereas, all were rated at Level One (HQR < 3.5) in flight. The differences in mean pilot ratings for each task between flight and simulation were significant at the 90 percent level of confidence. Clearly the pilots perceived that something was different in the simulation. The differences in pilot ratings, together with the pilots' supporting comments, imply that the fidelity of the simulated tasks was degraded.

Conclusions and Recommendations From Analyses of the Bob-Up Maneuvers

The comparative analyses, based on non-intrusive measurements of the piloting techniques and performance in the bob-ups, support the implication from the pilot ratings and comments that the fidelity of the simulated task was degraded. The comparative analyses, furthermore, help us to hypothesize what conditions might have caused the differences in pilot ratings and to suggest what remedies may improve the fidelity of the simulation.

The pilots' ratings, comments, characteristic lead compensation frequency, larger effective time delay, and lower closed-loop damping ratio in the simulator represent consistent evidence for their failure to employ motion feedback in the simulated bob-up in a way which complements visual feedback. Disparate motion and visual cues caused by the vertical motion washout, coupled with the visual CGI delay, are the likely underlying causes of the more lightly damped error reduction performance in the simulator and the pilot comments on the lack of heave damping in the simulator. The adopted control technique in the simulator may by necessity effectively ignore motion cues; such a conclusion is at least compatible with the evidence. The evidence therefore provides a compelling motivation to continue the investigation of the "full-motion" bob-ups in the simulator in order to test these causal hypotheses.

A cooperative effort has been made by Systems Technology, Inc., the National Aeronautics and Space Administration (NASA), and the U.S. Army Aviation Engineering Flight Activity, all working under direction from the U.S. Army Aeromechanics Laboratory, to obtain a better understanding of simulator hardware/software interactions and design requirements. One goal of this effort was to offer a method of "calibrating" pilot ratings and performance obtained in the Vertical Motion Simulator while pilots perform nap-of-the-earth tasks. This research has made considerable progress toward quantifying the terms of simulation validation, verification, and fidelity for a UH-60A Black Hawk rotorcraft. Results to date indicate that significant differences exist in the way that pilots execute a simple but aggressive bob-up maneuver in the Vertical Motion Simulator (VMS) when compared with actual flight. These qualitative and quantitative results indicate that several follow-on experiments need to be conducted in order to enable researchers to determine the best simulator configuration for simulation of rotorcraft in the VMS at Ames Research Center, NASA. Recommendations for these experiments are as follows.

- Perform the full-motion bob-ups in the simulator as planned.

- Motion washout algorithms should be designed to reduce the phase distortion of both the translational and rotational axes of the VMS, especially in the region of pilot crossover frequency for the piloting tasks under investigation.
- A dynamic image of the rotor tip path plane should be included in the visual scene.
- Improved visual scene content and detail in the ground plane are needed to improve the precision performance capability of the simulator pilot in nap-of-the-earth tasks.
- Phase lead compensation algorithms which minimize gain distortion are needed to reduce phase lags contributed by the visual CGI delay in the region of pilot's crossover frequencies for the piloting tasks.

The computer-generated visual image delay contributes to the pilot-induced oscillation in roll during bob-ups in the simulator. The visual delay, while important and always in need of reduction to further improve pilot performance, does not seem to be the sole reason for the documented difference in vertical translation performance between flight and simulated bob-ups. Likewise, the UH-60A mathematical model does not in itself seem responsible for the disparity between bob-ups. To understand fully and document whether or not the phase disparities between the motion and visual cues are the primary causes, as is now suspected, will require completion of the analysis of the other maneuvers--the hover turn, dash/quickstop, dolphin, and sidestep--as well as further analytical modeling and experimentation, which are already planned. Results from the analyses-in-progress of the hover turn not only support but also reinforce the conclusions from analyses of the bob-up maneuver.

Acknowledgements

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FLIGHT TESTS WITH A NEW HELICOPTER FORCE FEEL SYSTEM (FFS)

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SUMMARY

In a recent experimental program MBB Product Group Helicopters has investigated an artificial force feel system for the control of the pitch and roll axes of a Bo 105 helicopter. This work was carried out in cooperation with the DFVLR at Braunschweig using a special Bo 105 S3 equipped with a fly by wire control system. The safety pilot uses a mechanical control system. Goals of the program were to investigate if the use of such an artifical force feel system for the helicopter control would increase flight safety, allow a more precise control of a flight path, increase stability and reduce pilots workload.

Thus an experimental flight worthy system using two electro-hydraulik actuators for the pitch and roll axis respectively was implemented in the "flying simulator" Bo 105-S3.

A number of different force control laws were derived and flight tested. Although some contributions to the total control force were rated very differently by the various test pilots (e.g. contributions from the normal acceleration), as a general trend very favourable ratings were obtained. Two examples of these favourable ratings for highly dynamic manoeuvres (slalom and delphin) are an improved stability of the trim state and an improved flight path control in general.

Finally an outlook for the application of such a system in the flight control of future helicopters is given.

1. INTRODUCTION

Although numerous experiments concerning various control force feel systems (FFS) have been carried out for fixed wing aircraft, in particular for fighter aircraft, only very limited information on FFS's is available for helicopters (e.g. ref. 2. and 3.).

Thus a flight worthy experimental FFS was designed and installed in a Bo 105 operated by the german research organisation DFVLR at Braunschweig.

At the onset of the program a number of questions were posed for which it was tried to find at least some answers during the program.

Among the areas of interest were

- what kind of improvements can be obtained for the helicopter flight control using an FFS
- is the flight safety improved
- does an FFS allow a more precise flight path control, in particular for highly dynamic manoeuvres
- is the stability around the trimm condition improved
- is the pilot's work load reduced
- how are the various terms contributing to the total control force rated by the test pilots.

As can be seen from the questions already, most of the answers must rely to a large extend on subjective pilot ratings, unless very extensive and cumbersome evaluation methods are applied, e.g. to determine the reduction in pilot work load. So most of the results presented later were derived from pilot evaluations during the actual tests and in briefings after the tests.

2. DESCRIPTION OF THE EXPERIMENTAL SYSTEM

2.1 TEST HELICOPTER BO 105-S3

The helicopter used in the flight tests is a Bo 105 specially modified with a simplex "fly by wire" control system allowing this helicopter to be used as a flying simulator. In this helicopter the simulation pilot is located centrally in the front, whereas the safety pilot with a conventional mechanical control link is operating from the left hand side shifted to the rear. Although this particular feature of the test helicopter was not a prerequisite for the testing of the FFS, the installation and the tests instrumentation facilitated the experimental installation of the FFS considerably. Fig. 2.1 gives the control arrangement schematic for the pitch axis of the test helicopter. The same arrangement is used for all four control axes of this helicopter. For the test purposes the standard electro-mechanical trim motors and spring assemblies were replaced by electro-hydraulik actuators in the pitch and roll axes. This portion of the pitch axis is marked by a circle in fig. 2.1. The electro-hydraulic actuators were chosen, because they allowed to vary the spring constants and variations of control forces in a wide range by just changing the electronics. This arrangement also possessed a high margin of safety in case of an FFS electronics failure. For one the elctro hydraulik actuators had a by-pass feature and in addition of course the safety pilot, who was participating during all flight tests, could take over in cases of real trouble. No such instance occured however during the flight tests.

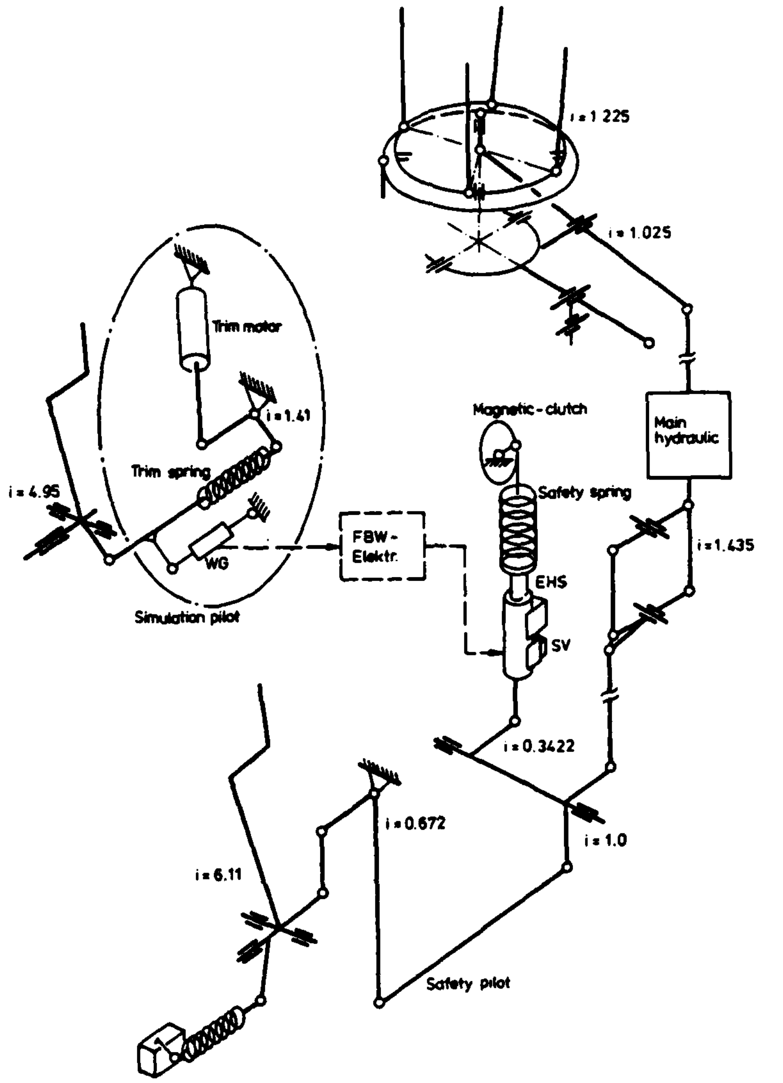


Fig. 2.1 Schematic of the Control arrangment for the simulation and safety pilot of the Bo 105-S3 (example pitch axis)

2.2 EXPERIMENTAL FFS

A schematic outline of the experimental FFS is shown in Fig. 2.2. Again the electro-hydraulik actuator is shown for the pitch axis only. The flight state sensors and the corresponding parameters which could be used to form the control force laws are shown to the right of the FFS computer. The initial design of the system allowed the following parameter combinations for the two axes:

Pitch axis

- control input
- control input rate
- pitch rate x airspeed/or normal acceleration

Roll axis

- control input
- control input rate
- roll rate x airspeed

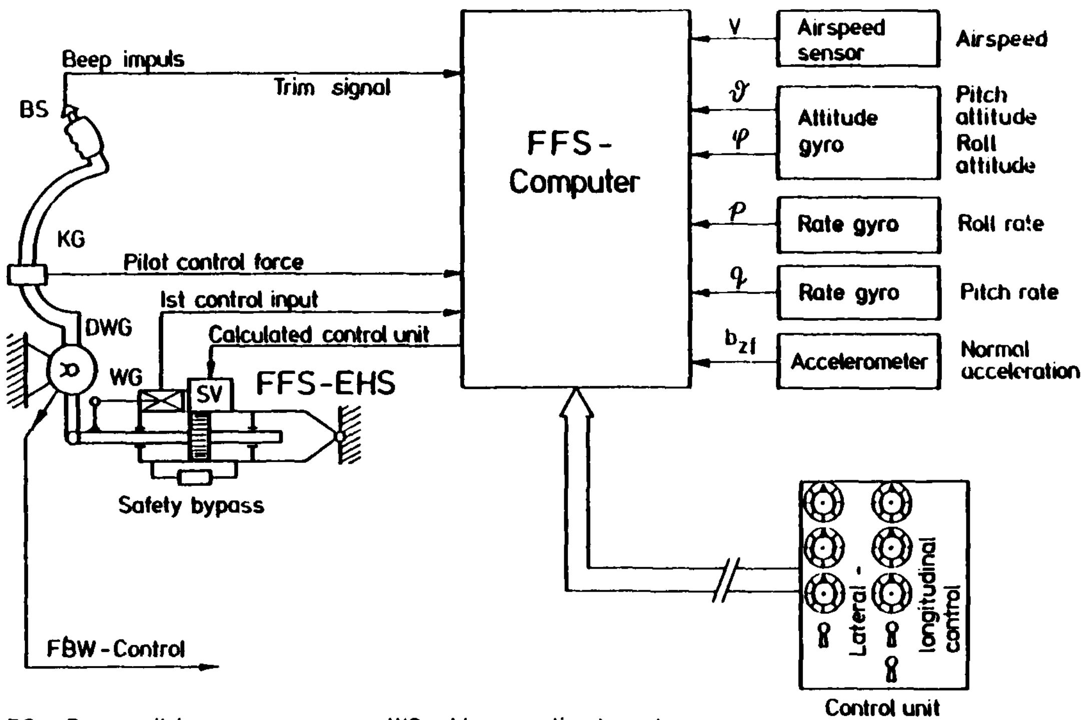
During the tests it was found that it was beneficial to use an additional term

attitude x airspeed

for the pitch and roll axis respectively.

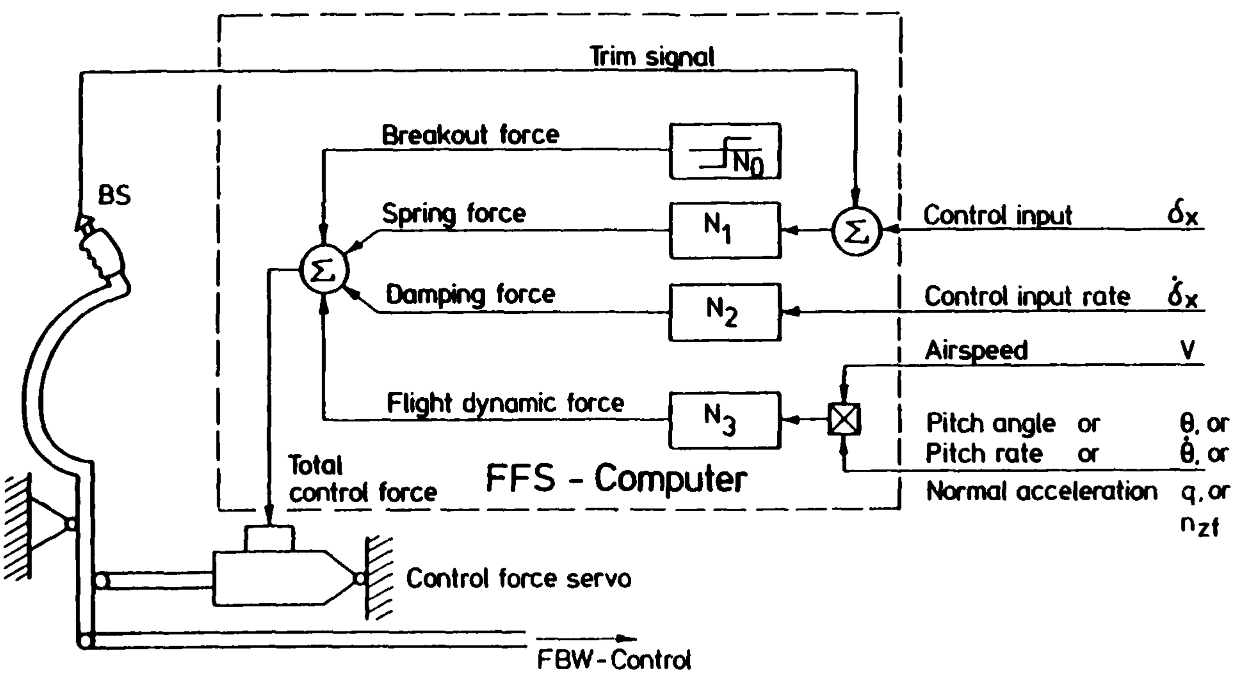
The various terms and their possible combinations are shown in Fig. 2.3. Again the pitch axis is shown only, since the terms for the roll axis were quite similar, except that the normal acceleration was missing.

The hardware components of the experimental FFS are shown in Fig. 2.4. Besides the two electro-hydraulik actuators with servo valves, the electronic unit (computer), the control unit and the modified stick with a force sensor are shown. These electro-hydraulik actuators were used to replace the trim motor and spring assemblies for the pitch and roll axis respectively as indicated in Fig. 2.1.



BS Beep switch
KG Pilot control force sensor
DWG Angular motion transducer
FBW Fly-by-wire
WG Linear motion transducer
SV Servo valve
EHS Electro hydraulic actuator

Fig. 2.2 Schematic Outline of the experimental Force Feel System (FFS) for the investigation of different control laws



BS Beep Switch
(Σ) Summing point
N Control force coefficient
⊠ Multiplier

Fig. 2.3 Control Force Contributions for the Pitch Axis

Fig. 2.4 Hardware components of the experimental FFS

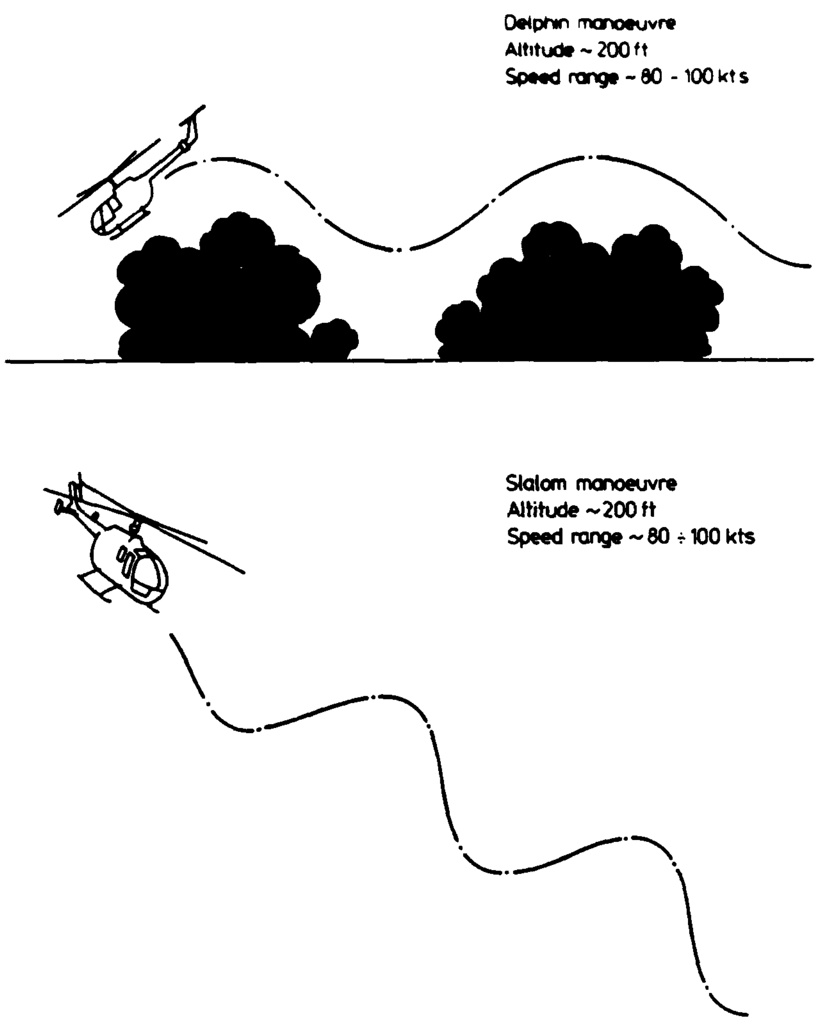


Fig. 2.5 Schematic representation of the "Delphin" and "Slalom" manoeuvres

2.3 FLIGHT MANOEUVRES AND INSTRUMENTATION

Most of the actual evalutations were done for two highly dynamic manoeuvres called "delphin" and "slalom". These manoeuvres were chosen being representative of so called tactical NOE-flight. Since the simulation "fly by wire" control system was only a simplex system, the manoeuvres were flown in an altitude above 200', which was considered the safety altitude. Most of these manoeuvre tests were flown in the speed range of 80 - 100 kts. A schematic representation of the "delphin" and "slalom" manoeuvres are given in Fig. 2.5. Both manoeuvres require rapid and extensive control inputs and were thus well suited to investigate the influence of the various control force contributions.

A schematic of the potentiometers used for parameter variations and the parameters recorded on the test equipment are shown in Fig. 2.6. Most of the data was recorded on an airborne magnetic tape as well as transmitted directly to the ground station, where the important data could be observed directly on two 8 channel data recorders during the actual tests. Most flight tests were initially started with some basic values of the control force parameters. Any chamges of this values were announced by the pilot to the ground station and noted down in the flight log. Any comments of the pilots during the tests were also recorded and later used for the evaluation of the parameters. Five different test pilots carried out a total number of 16 flight tests. A summary of all test results is presented in the following chapter.

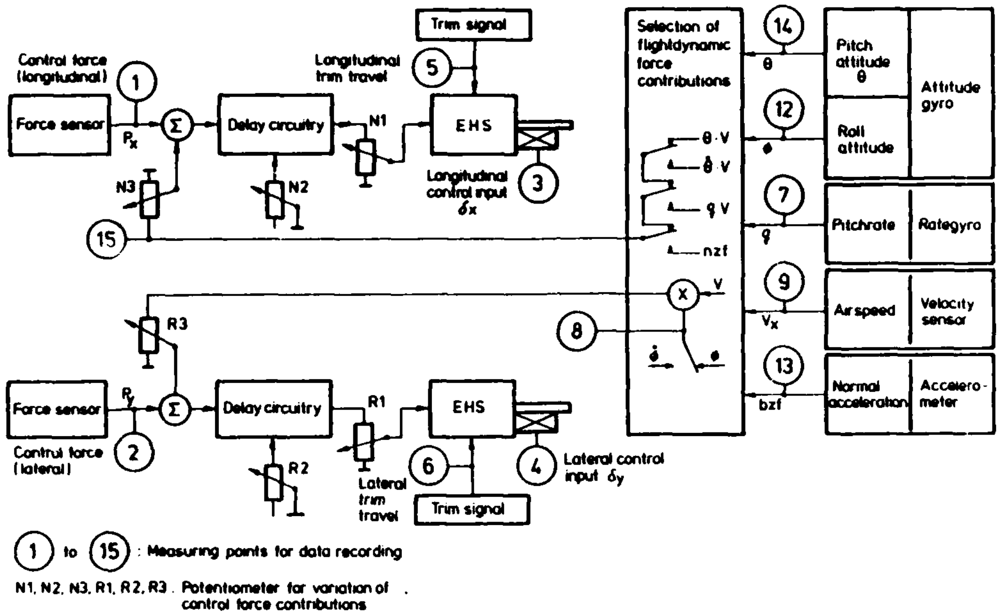


Fig. 2.6 Schematic of potentiometers and mesured parameters

3. FLIGHT TEST RESULTS

3.1 FLIGHT MANOEUVRES

The highly dynamic flight manoeuvres slalom and delphin flight, as investigated during the tests described, can be approximated by rather simple sin-functions.
This also holds true for the control motions.

For the delphin flight the functions are

$$\theta = 15° \sin \frac{2\pi}{7} \cdot t \text{ [sec]}$$

$$\delta_x = 27 \text{ [mm]} \cdot \sin \frac{2\pi}{7} \cdot t \text{ [sec]}$$

and for the slalom flight

$$\phi = 42° \sin \frac{2\pi}{9} \cdot t \text{ [sec]}$$

$$\delta_y = 44 \text{ [mm]} \sin \frac{2\pi}{9} \cdot t \text{ [sec]}$$

Table 3.1 shows the average values and the standard deviation for flight dynamic- and control parameters as evaluated from the flight test data for the slalom and delphin manoeuvre.

Delphin-Flight			mean values (arithmetic)	standard deviation
Pitch attitude change	$\Delta\theta$	(°)	14,5°	2,2
Period	T	(sec)	7,2	2,1
Normal acceleration	n_{zf}	(m/sec²)	10,9	
Pitch rate	$\dot{\theta}$	(°/sec)	13,6	2,8
Longitudinal Control Input	$\Delta\delta_x$	(mm)	26,8	6,2
Maximum longitudinal Control input	$\Delta\delta_{x,max}$	(mm)	272	
Selection of delphin-flights				
Control input	$\Delta\delta_x$	(mm)	30,2	4,8
Control input rate	$\dot{\delta}_x$	(mm/sec)	32,7	
Period	T	(sec)	5,8	0,4
Longitudinal Control force	P_{Px}	(N)	15,8	3,6
Longitudinal Control force rate	$\dot{P}_{Px}$	(N/sec)	17,1	
Stalom-Flight				
Roll attitude change	$\Delta\phi$	(°)	41,5	6
Period	T	(sec)	8,6	1,5
Roll rate	$\dot{\phi}$	(°/sec)	36,4	4,3
Lateral control input	$\Delta\delta_y$	(mm)	44,1	8,6
Maximum lateral control input	$\Delta\delta_{y,max}$	(mm)	195	
Control input rate	$\dot{\delta}_y$	(mm/sec)	31,9	
Lateral control force	P_{Py}	(N)	11,9	4,1
Lateral control force rate	$\dot{P}_{Py}$	(N/sec)	12,2	

Table 3.1. Summary of Flightdynamic- and Control-Parameters

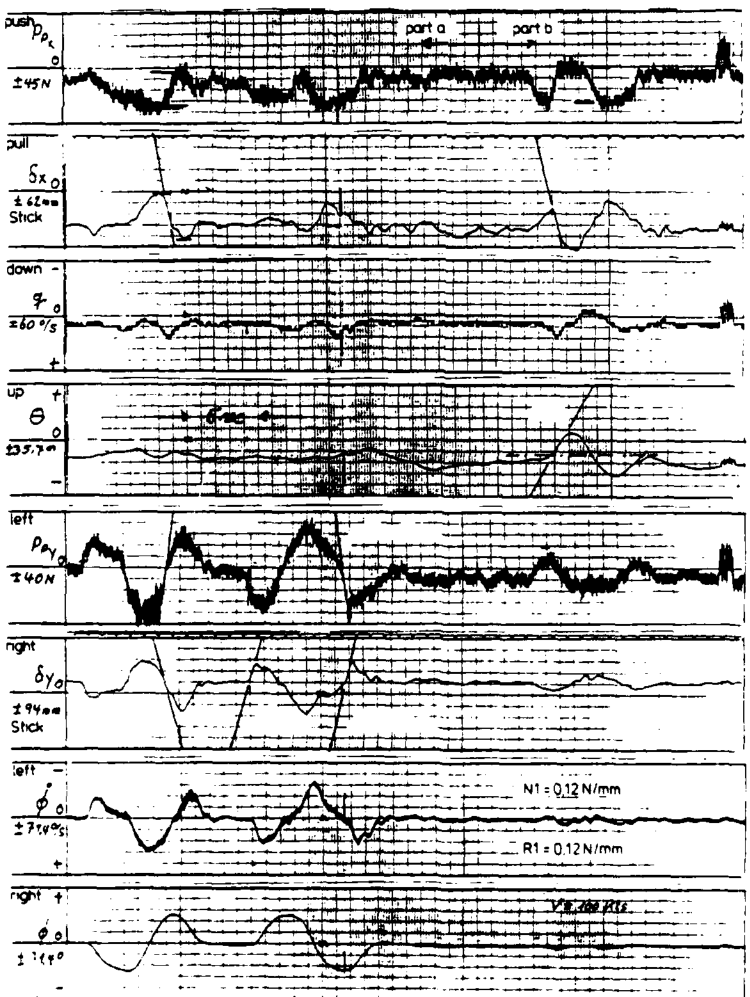


Fig. 3.1 Flight No. 1 Part a: Slalom manoeuvre
Part b: Delphin manoeuvre

The rather large control rates were not anticipated at the beginning of the flight tests and let to a change of the servo values of the electro-hydraulik actuators allowing these values. The values shown were calculated from data recordings, two examples of which are shown in Figs. 3.1 and 3.2. From the different traces it can be clearly seen, that there are also longitudinal force contributions during slalom flight and lateral force contributions during delphin flight due to the cross coupling of the control responses.

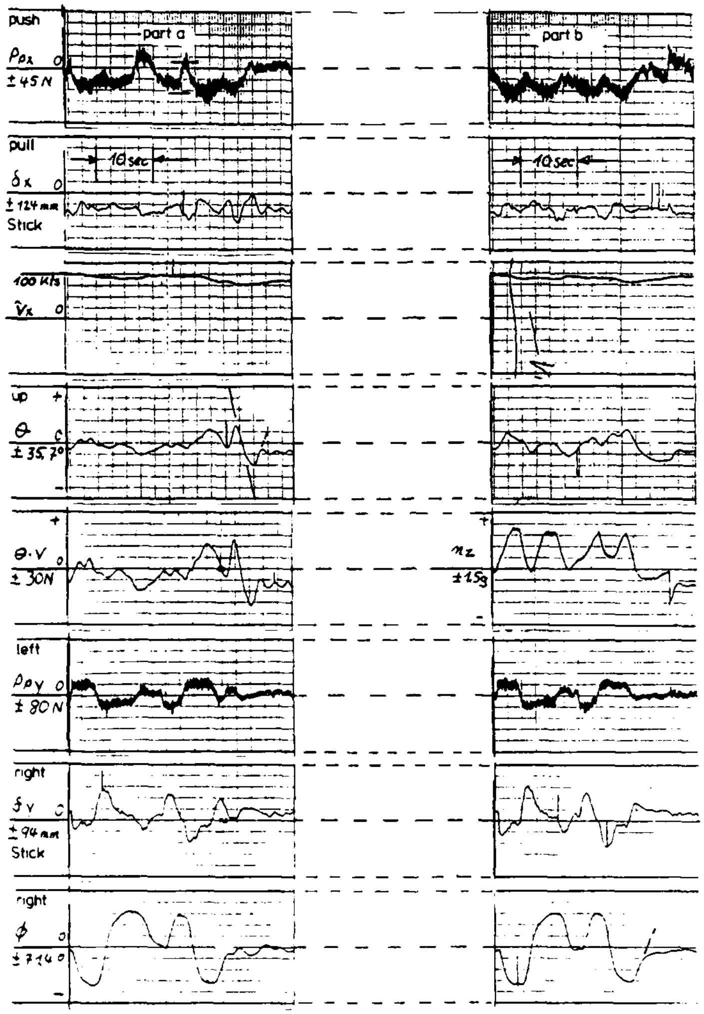


Fig. 3.2 Flight No. 16 Part a: Slalom manoeuvre
Part b: Delphin manoeuvre

3.2 CONTROL FORCE COEFFICIENTS

Taking into account the subjective ratings of five different test pilots good agreement was obtained in general about the terms and the magnitude of the corresponding coefficients contributing to the total control force. A contradictory judgement was only obtained for the normal acceleration term, which was rated from adequate to useless.

The following table 3.2 summarizes the various control force coefficients according to the subjective pilot ratings. The table starts with the break out force values, which were actually not truly optimized, because the values selected, are the lowest possible with the kinematic design of the Bo 105 controls during manoeuvre flight.

Comparing the spring force values with the values for the standard Bo 105 and BK 117 or in particular with values taken from reference 2, which were of course obtained for a much bigger helicopter, shows, that there are considerable differences. These differences become even more pronounced when looking at the damping force portions. The Bo 105 pilots were unanimously rather sensitive to this contribution and have selected it to be rather small in order to have no interference with swift control rates.

Also when comparing the flight dynamic term in line 5, $q \cdot V$ and the load factor terms n_{zf} in line 6 with the value from ref. 2, which are corresponding contributions indicating the flight loads on possible critical components, a similar difference becomes apparent. The values for the S-67 being considerably higher. By evaluating these differences it should also be kept in mind, that the values for the S-67 were obtained for normal cruise flight and not for highly dynamic manoeuvres.

The results of the Bo 105 flight tests indicate, that the force coefficients are chosen by the pilots such, that the maximum values of the different contributions to the total control force lie in the average between 5 and 10 N. It should be kept in mind though, that these different contributions must not simply be added to obtain the maximum control forces, since the maximum values of the different contributions do not occur simultaneously.

		BO 105-FFS delphin&slatom flight Pitch	Roll	BO 105 Standard Pitch	Roll	BK 117 Standard Pitch	Roll	S-67/FAS Ref.2 Cruise P.tch	Roll
1. BREAKOUT FORCE	(N)	$N_0 = 5^*$	$R_0 = 5^*$	5	5	5	5	8,9	4,9
2. SPRING FORCE	(N/mm)	$N_1 = 0{,}1 \div 0{,}15$	$R_1 = 0{,}1 \div 0{,}2$	0,23	0,27	0,24	0,28	0,88	0,09
3. DAMPING FORCE	(N/mm/sec)	$N_2 = 0{,}03$	$R_2 = 0{,}03$					0,53	0,35
4. $\Theta \cdot V$ RESP. $\Phi \cdot V$	(N/°m/sec)	$N_3 = 0{,}008$	$R_3 = 0{,}003$						
5. $q \cdot V$ RESP. $\dot{\Phi} \cdot V$	(N/m/sec²)	$N_3 = 0{,}6$	$R_3 = 0{,}6$						
6. n_{zf}	(N/m/sec²)	$N_3 = 0{,}6$						9,1	

* smaller values are not possible for dynamic manoeuvers due to the kinematic design of the controls

Table 3.2 Control Force Coefficients as rated by pilots

4. BK 117 FORCE FEEL SYSTEM (FFS)

A simple version of a pitch axis force feel system has been applied on the new BK 117 helicopter as a derivate from these investigations. Since hydraulik power controls are installed in cyclic and collective main rotor control axes, artificial force feel is included in both cyclic axes in the form of a bidirectional preloaded spring box and an electric trim motor. After normal pilot's control inputs, the cyclic control forces may be cancelled by operating a conventional "beep" trim switch situated on the cyclic grip. The pilot retains pressure on the switch until the output of the trim motor is such to recentre the spring box and remove the control force.

For flight under IFR, the conventional trim system is modified as shown in Fig. 4.1 to enhance piloting qualities. In the switch position "FFS ON", the cyclic "beep" trim switch can start, in the trim signal generator, a digitally build integrator, the integration rate of which is controlled by the "Trim speed" switch. The output side of the integrator commands the trim motor position which is measured by the "pick-off", and the deviation signal operates the power amplifier which dirves the trim motor to the commanded position. Thus the cyclic "beep" switch controls the Position of the trim motor in the conventional way.

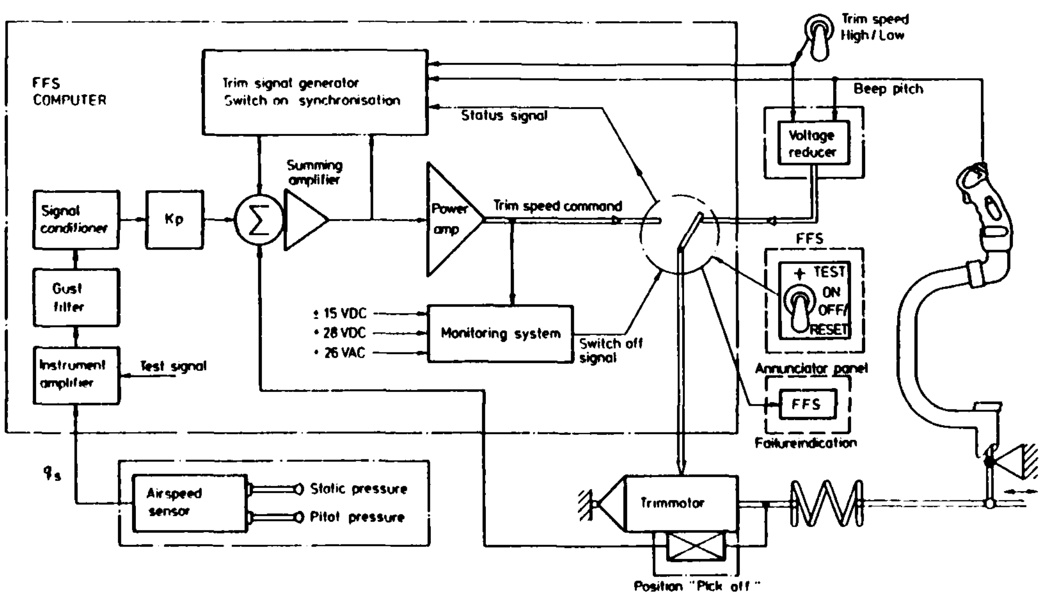


Fig. 4.1 Force Feel System BK 117

The trim motor is also influenced by the helicopter flight state. Dynamic pressure (q_s) is measured by a pitot static airspeed sensor. The electric signal is conditioned by a low-pass filter which eliminates gusts and high frequency transients of 0.2 Hz and above. Furthermore sensed dynamic pressures equivalent to about 20 kts and below are also removed in the electronic control unit. The signal is amplified by the gain constant Kp, converted into a control signal for the trim motor and summed to the "beep" trim command. The handling characteristics of the FFS are such that, following a pilot's cyclic input to cause an airspeed change (without retrimming) the trim motor position is modified. Thus the longitudinal cyclic control force derived by the stick position stability is enhanced by the additional dynamic pressure signal which is a function of airspeed. The selection of the gain factor Kp has been such to optimise the cyclic force handling qualities and to fulfill the IFR requirements for static longitudinal control force stability. Following a change of flight state involving an airspeed change, the control forces can be trimmed out in the conventional manner using the "beep" trim switch to establish a new trim command datum.

The normal configuration for IFR is with the FFS SWITCH in the ON position. The pilot may, however, revert to conventional "beep" trim alone (if desired during VFR flight, or in the event of a malfunction) by selecting "FFS OFF". In this state the trim signal generator goes into a follow up mode to ensure synchronisation when turning "FFS ON".

The output of the "Power Amplifier" is monitored (Figure 4.2). An output voltage of more than ± 19 V for longer than 0.2 sec is defined as a fault. This is equivalent to a stick movement of approx. 5 mm. When a fault occurs, the monitoring system switches from FFS to normal "beep" trim and the lamp FFS in the annunciator panel is illuminated.

The external voltage supplies, which are 26 V AC, 400 Hz and 28 V DC and the internal voltages + and - 15 V DC, are also monitored to determine whether they are within the specified ranges. The FFS is switched off, when one of the voltages is out of the specified bandwidth.

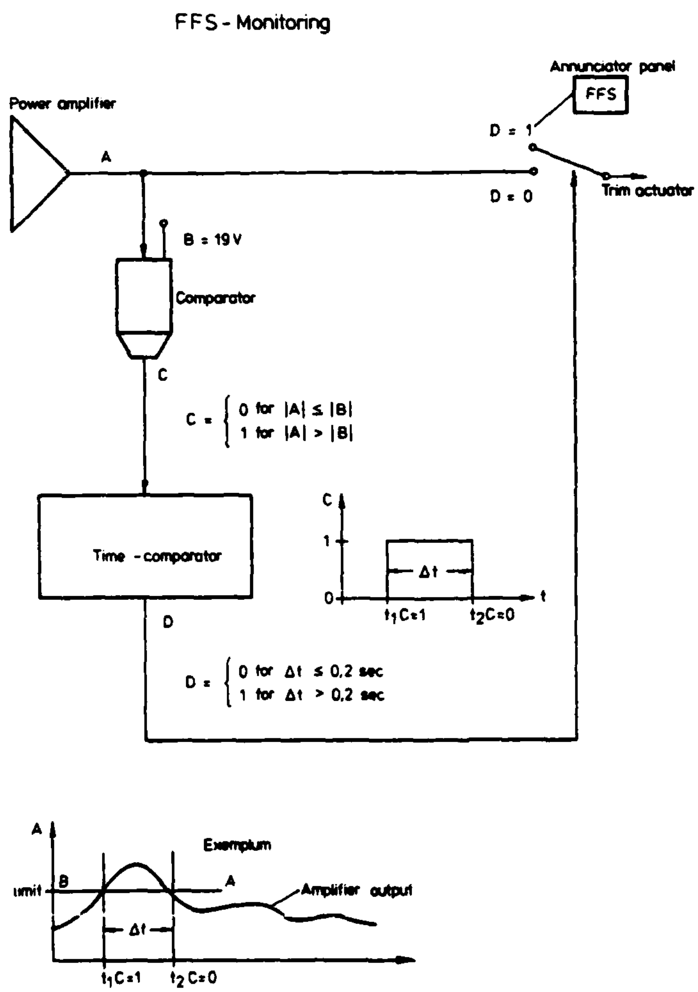


Fig. 4.2 Monitoring of trim speed command

5. CONCLUSION

From flight tests with an experimental two axes (pitch and roll) force feel system on a Bo 105 helicopter the following conclusions can be drawn for highly dynamic manoeuvre flight:

- Control motion intensity during highly dynamic manoeuvres (slalom and delphin) is higher than initially anticipated.
- During delphin manoeuvres normal accelerations in the range of $0 \div 2$ g occured
- Spring rate contributions should be smaller than on the standard Bo 105 (approx. half the value).
- Damping force contributions resulting from control rates should be suppressed as much as possible.
- Introduction of additional flight dynamic terms like $\Delta\Theta \cdot V$, $\Delta\phi \cdot V$ and $q \cdot V$ or $\dot{\phi} \cdot V$ were rated favourably by the test pilots.
- The latter terms increase the stability of the helicopter around the trim state.
- The rating for the normal acceleration term n_{zf} was somewhat contradictary between different test pilots ranging from useful to useless.
- The general trend of the pilot ratings confirm that the FFS
  - allows a more precise flight path control
  - increases stability
  - reduces pilot's workload.

The experimental FFS used in the flight tests did not allow the combination of several of the flight dynamic terms as they were determined useful during the tests. In future tests a modified FFS will also allow to investigate as well such combinations as functions depending on the velocity for the $\Theta \cdot V$ and $\phi \cdot V$ terms. In addition simplified versions using the mostly available spring trim motor combinations similar to the Bk 117 system should be investigated.

The well rated parameters combinations than also need to be verified for other more normal flight manoeuvres like hover, cruise and approach.

6. NOMENCLATURE

N_1	FFS-Parameter for the longitudinal control input
R_1	FFS-Parameter for the lateral control input
N_2	FFS-Parameter for the longitudinal control input rate
R_2	FFS-Parameter for the lateral control input rate
N_{3q}	FFS-Parameter for the Implementation of $q \cdot V_x$ or n_{zf}, on the longitudinal control
$N_{3\Theta}$	FFS-Parameter for the implementation of $\Theta \cdot V_x$ on the longitudinal control
$R_{3\dot{\phi}}$	FFS-Parameter for the implementation of $\dot{\phi} \cdot V_x$ on the lateral control
$R_{3\phi}$	FFS-Parameter for the implementation of $\phi \cdot V_x$ on the lateral control
P_{Px}	longitudinal control force
P_{Py}	lateral control force
P_A	breakout force
δ_x	longitudinal control input
δ_y	lateral control input
ΔP_{Px}	amplitude of longitudinal control force
ΔP_{Py}	amplitude of lateral control force
P_{Pxmax}	Maximum longitudinal control force
ΔP_{Pxm}	average amplitude of longitudinal control force ΔP_{Px}
ΔP_{Pym}	average amplitude of lateral control force ΔP_{Py}
$\Delta\delta x$	amplitude of longitudinal control input
$\Delta\delta y$	amplitude of lateral control input
n_{zf}	normal acceleration
ϕ	roll attitude
Θ	pitch attitude
V_x	airspeed (pilot-static)
p	roll rate (with respect to helicopter-axes)
q	pitch rate (with respect to helicopter axes)
HS	helicopter
FFS	force feel system
NOE	nap of the earth
EHS	electro-hydraulic actuator

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HELICOPTER ACTIVE CONTROL WITH ROTOR BLADES RELAXATION

by

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SUMMARY

An active modal control for high performances combat helicopters in forward flight is presented in this study. A Gust Alleviation Control (G.A.C.) strategy based on the spectral data computed from the flexible blade structural mode of vibration measurements, is employed to relax, by appropiate longitudinal cyclic pitch modulation, the flatwise bending moments induced by environmental disturbances. The restoring cyclic pitch commands are derived processing the output data from an electro-optical laser sensor (L.S.U.) by means of a microprocessor performing the Spectral Power Density (S.P.D.) real time computations; these data, obtained implementing a Fast Fourier Transform algorithm and observed within a frequency window centered at the first bending mode frequency, are employed as a measure of the actual vibrational level existing on the blade. To reduce the helicopter rigid response sensitivity to the G.A.C. system actuations, its driving signals are applied to the Longitudinal Pitch Decoupling (L.P.D.) unit making the helicopter attitude and vertical velocity component decoupled, the last one regulated by the collective pitch channel through a radioaltimeter closure. The effectiveness of the G.A.C. system is investigated by digital simulations. The obtained results yield promising indications regarding the G.A.C. system feasibility and its potential usefulness particurarly in high speed low level combat operations, are pointed out.

LIST OF ABBREVIATIONS/ACRONYSMS

A	State matrix
B	Control matrix
C	Output matrix
C.Y.P.D.	Collective pitch decoupled channel
C.Y.P.D.	Cyclic pitch decoupled channel
d_c	cyclic pitch command
d_{cc}	collective pitch command
D.F.T.	Discrete Fourier Transform
E	Blade bending modulus of elasticity
E ( )	Mean value
F_k	Generalized force for the k-th bending mode
F.F.T.	Fast Fourier Transform
g	Blade displacement
G.A.C.	Gust Alleviation control
I	Blade section second moment of area, flatwise bending
I_b	Blade mass moment of inertia respect to flapping hinge
I.D.F.T.	Inverse Discrete Fourier Transform
I.F.F.T.	Inverse Fast Fourier Transform
I.M.V.S.	Inertial Measuring Velocity System
M_k	Generalized mass for the k-th bending mode
N	Number of discrete observations in F.F.T. computations
N_f	Number of points in the Barlett window width
K_s	Forward gain or gain matrix
K_f	Feedback gain vector or matrix
P	Spectral Power Density
q	Helicopter pitch rate

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r	Blade section distance from rotor hub
R	Rotor radius
S_c	Cyclic decoupled subsystem
S_{cc}	Collective decoupled subsystem
S.P.D.	Spectral Power Density
T_s	Blade rotational period
T_c	Sampling time
Tx	Synchro trasmitter
u	Long. Vel. Comp. in body axes
V	Helicopter airspeed
V.G.	Vertical Gyro
R.G.	Rate of gyro
w	Vert. Vel. Comp. in body axes
w (t)	Barlett window function
x	r/R

I - INTRODUCTION

Focusing the particular aerodynamical, structural and control problems inherent to the two different configurations, the maneuvre and gust alleviation active control strategy (Ref.1) applied to the fixed wing aircrafts, can be extended, in order to control the rotor blade flexibility, to the rotor wing vehicles. The current trend in the helicopter gust alleviation technology is oriented to counteract the full development of the gust induced airloads by pitching the helicopter or by rotating the rotor blades into the gust. This strategy requiring, for the blade load relaxation, an appropiate cyclic pitch control, is implemented in this study as an organized digital modal control system where a spectral computing unit plays an important role in the gust alleviation process.

The Individual Blade Control (I.B.C.) concepts, proposed by various authors and experimented by M.I.T. researchers (Ref. 2,3) is applied in formulating an advanced modal control strategy where an electro-optical laser (L.S.U.) sensor is employed to measure the blade linear displacements in respect to a reference plane. The cyclic pitch actuations involved in the Gust Alleviation Control (G.A.C.) are governed by a propotional and derivative feedback structure driven by a signal proportional to the actual vibrational energy existing on the blade. Extending to the conventional flying vehicles the modal technique (Ref.5,6) proposed for the large flexible structure in space shape control, the active control actuations operated by the G.A.C. system are regulated employing the Spectral Power Density (S.P.D.) data relative to the flexible blade mode of vibration. A sinusoidal forcing function, the magnitude of which is made proportional to the S.P.D. computed in a specified frequency window opened in the range of the dominant structural modes, is applied to the cyclic control channel; this driving signal will make the blades rotating in pitch with a generation of additional airloads relaxing the gust induced airloads supported by the blade. The associated rotor precessional effects arising from the rotating blade cyclic pitch actuation, will produce a rotor tip path plane tilt into the gust. The last effect, involving the helicopter attitude changes,will provide a direct gust effect compensation which is particularly effective in hingless rotors which are known to transfer large moments to the rotor hub.

To cope with the rigid operative constraints predicted for the class of helicopters treated in this study, the attitude variations due to the G.A.C. system actuations must not affect the helicopter flight path. For this reason the G.A.C. system driving signals are applied to the Longitudinal Pitch Decoupling (L.P.D.) unit described in Ref.4; this device is a multi-feedback control system processing all the observable helicopter state variables is such a way that, if a command signal is applied to its "cyclic pitch channel" input, only the helicopter attitude is varied not affecting the helicopter vertical velocity; instead, the last state variable can be reached, without affecting the helicopter attitude, by a command signal applied to the "collective pitch channel". Serving the L.P.D. collective channel with a reference datum derived by an Inertial Platform or a Radioaltimeter, the G.A.C. system will be made capable, within the validity limit of the linear state variables decoupling theory, to regulate the helicopter attitude, as required to develop the gust alleviation strategy, without affecting the helicopter flight path.

In this preliminary G.A.C. system feassibility investigation tending essentially to obtain, with a reasonable engineering accuracy, the basic informations on the proposed control system structure and, in first approximation, the fundamental characteristics of the G.A.C. dynamical behaviour, the mathematical models for the flexible blade bending

degree of freedom and for the rigid helicopter longitudinal dynamics are assumed linear. Additional blade degree of freedom, such as the in-plane bending and twisting, which are significant particularly for the hingeless rotors, will be treated, as extensions in the area of the G.A.C. system investigations, in next future work. The present analysis,which is referred to a specified value of the blade hinge offset, can be extended to a different value of this parameter to simulate a particular blade "degree of rigidity" as required in the design.

In the Section I, the helicopter rotor configuration and the blade geometrical and inertial characteristics assumed as an introductive explanatory model for this presentation, are indicated. The results of the modal analysis for the blade lumped mass model are presented in Section 2. The linear mathematical model for the blade structural model describing its first flatwise bending mode is discussed in Section 3 and in the next section this model is employed to derive the feedback control structure providing the necessary damping into the vibratory blade dynamics. The spectral modal control concepts and their implications in computing the S.P.D. of the blade displacement and velocity time functions observed by the the electro-optical laser sensor are treated in Section 5; here the S.P.D. computing aspects involving a real time, high speed dedicated F.F.T. microprocessor and some software observations regarding the routine employed to generate a frequency window through which the S.P.D. is evaluated, are discussed. In Section 6 follows the description of the general configuration of the G.A.C. system including the L.P.D. unit. The results of the digital simulations considering the dynamical behaviour of the bare and G.A.C. controlled blade subjected to a gust excitation and the correspondent dynamical effects on the rigid helicopter are presented and discussed in the last concluding section.

2 - ROTOR AND BLADE CHARACTERISTICS

The geometrical, inertial and aerodynamical characteristics of the untwisted, untapered blade with uniform mass and stiffness radial distribution, the rotor configuration and its reference operating condition assumed in this study are indicated in Table I.

3 - BLADE MODAL ANALYSIS RESULTS

The modal behaviour of a non rotating uniform single blade represents a fundamental step in the study of the flexibility effects on the helicopter dynamics. Since the investigation is essentially devoted to solve the vehicle control problem under environmental disturbances, only the fundamental degree of freedom influencing the helicopter stability and controllability, i.e. the flatwise bending mode, is considered in this study. To make the analysis essential in terms of engineering accuracy requested to analyze the effects of the flexible blade motions on the helicopter dynamics, a lumped mass structural model, where the blade structure number of rigid masses connected by weigthless connectors having the same elastic property of the actual physical structure, has been considered for the blade modal analysis. In terms of normal flatwise bending modes, the solution of the eigenmode problem given by a series expression:

$$z = \sum_{x=1}^{\infty} \eta_k \quad (r) \quad g_k \quad (t) \qquad (I)$$

where η_k (r) is the mode shape for the k-th mode referred to the blade section at the distance r from the hub hinge and g_k (t) is the blade displacement time function relative to the k-th bending mode. The results of the eigenvalue and eigenmodes problems relative to the non rotating blade modelled as a 5-masses lumped model with zero structural damping are given in Table 2 in which the modal shape relative to the first flatwise bending mode normalized in respect to the blade tip section displacement is described.

The linear representation of the k-th bending mode for the non rotating undamped blade is given by the second order differential equation:

$$M \ddot{g}_k \ (t) + K \ g_k(t) = F \ (t) \qquad (2)$$

where M and K are respectively the mass and stifness matrices and F is the vector of the generalized forces associated with the k-th bending mode. The modes relative to the non rotating blade are assumed as a reasonable approximations for the rotating blade modes. In the next section the generalized aerodynamic forcing function causing the blade bending in a centrifugal force field is determined and the blade bending equation relative to the first flatwise bending mode is given.

TABLE I - Blade and rotor characteristics and Kinematics

Datum	Symbol	Dim.	Value
Airfoil	-	-	NACA 00112
Chord	c	m.	0.558
Lift Slope	c_{p_α}	I/rad	5.73
Blade weight per unit length	w	N/m	100.4374
Equivalent hinge offset	e_h	m.	0.142
Flapwise bending stiffness	E I	N.m^2	0.860268 10^5
Flapwise mass Moment	I_b	N.m.	10.9916
Blade Radius	R	m.	7.1092
Blade Number	N_p	-	3
Lock Number		-	5.79
Rotational speed	N_R	R.P.M.	360
Rotational frequency		rad/sec	37.68

TABLE 2 - RESULTS OF THE MODAL ANALYSIS FOR THE NON ROTATING
5 - LUMPED MASSES UNDAMPED BLADE MODEL

Natural frequencies		Modal Shape (ω_1 = 31.5)	
Mode (k)	Freq. (rad/sec)	x=r/R	η(k)
1	31.5	0	0
2	212.2	0.2	0.059886
3	382.2	0.4	0.219028
4	601.6	0.6	0.44659
5	924.1	0.8	0.71643

3 - ELASTIC BLADE DYNAMICAL MODEL

The mathematical model for the rotating elastic blade associated with its flatwise bending mode is formulated on the basis of the theory given in Ref. 2. The differential equation for the blade bending out of the rotational plane in terms of blade normal mode is expressed by:

$$\bar{M}_I \ \ddot{g}\ (t) + \bar{D}_I \ \dot{g}\ (t) + \bar{C}_I \ g(t) = \bar{F}_I \tag{3}$$

where $g(t)$ is the generalized blade displacement defined in (I) and specifically referred to the first bending mode, $\bar{M}_I$ is the generalized mass defined:

$$\bar{M}_I = \frac{M_I}{I_b} = \int_0^I \left(\eta_I\ (x)\ /\ R\right)^2 \ R^3 \ dx \tag{4}$$

$\bar{D}_I$ is the generalized viscous damping due to the additional airloads induced by the blade angle of attack changes in a quasi-static aerodynamic field involved in oscillating airfoil in incompressible flow:

$$\bar{D}_I = \frac{\gamma\Omega}{2}\int_0^I \left(\eta_I (x) / R\right)^2 x \, dx \tag{5}$$

$\bar{C}_I$ is the generalized spring constant defined:

$$\bar{C}_I = \bar{M}_I \, \omega_I^{\,2} \tag{6}$$

$\bar{F}_I$ is the generalized force due to the airloads acting on the blade due to the blade pitch angle (ϑ) changes:

$$\bar{F}_I = \frac{\gamma\Omega^2}{2\bar{M}} \int_0^I \left(\eta_I (x) / R\right) x^2 \, dx \tag{7}$$

The blade flatwise bending equation (3) expressed in phase variable form becomes:

$$\dot{\underline{x}} (t) = A_b \, \underline{x} (t) + B_b \, \vartheta (t) \tag{8}$$

where the state vector $\underline{x}$ (t) is defined:

$$\underline{x} (t) = g(t), \dot{g} (t) \tag{9}$$

The equation (8) will be used, as a first approximation for the system control analysis purposes, for modelling the elastic blade.
The characteristic equation for the system (8) referred to the case at hand will be:

$$D(s) = s I - A = s^2 + 0.260535 \, s + 964.10.25 =$$
$$= (s \lambda + {}_I) \, (s + \lambda_2)$$
$$\lambda_{I,2} = -0.1302675 \pm j \, 31.049$$

yielding a pair of complex conjugate roots indicating a very small relative dampling (ξ = 0.004272) and a natural frequency which is almost coincident to the first bending mode natural frequency.

4 - BLADE AUGMENTATION CONTROL SYSTEM

To provide an adequate damping to the blade first bending mode, the longitudinal cyclic pitch control is designed as a feedback structure implementing a proportional plus derivative control law based on the blade displacement and velocity measurements performed by an electro-optical sensor. The blade pitch actuations are governed by a servomotor with a dominant time constant of 0.004 sec. Augmenting the state vector $\underline{x}$, describing the flexible blade dynamics (8), with the servo displecement state variable (δ) defining the first order equivalent servo dynamic, the augmented state equation for the servo blade system will be expressed by:

$$\dot{\underline{x}} (t) = A_{bs} \, \underline{x} (t) + B_{bs} \, i (t) \tag{10}$$

$$\underline{x} (t) = \left(g(t), \; \dot{g} (t), \; \delta (t)\right) \tag{11}$$

where i (t) is the servomotor forcing function. The system (10) is enclosed in a feedback branches involving all the elements of the state variable $\underline{x}$ (t) through feedback gains regulated to obtain a substantial damping increase in the blade primary variable g(t). The control law employed for this purpose is expressed by:

$$i(t) = K_s \left(i_{ref} (t) + K_f^T \, \underline{x} (t)\right) \tag{12}$$

Regulating the feedback vector gain (K_f) and the forward gain (K_s) as follows:

$$K_s = 17.3935$$
$$K_f = \left(K_{f_\alpha} , K_{f_{\dot{\alpha}}} , K_{f_\delta}\right) = \left(0.3054 \,\vdots\, 0.01592 \,\vdots\, 1.22 \; 10^{-3}\right)$$

the blade damping will be increased to the value of 0.5 and the ratio of the blade rotational frequency and the first bending natural frequency is increased to the value of I.IO yielding a favourable effects on matching the rotor helicopter flight dynamics. Aggregating the servo blade state equation (10) to the control law (12), the damped blade - servo system can be expressed by a closed loop state equation:

$$\dot{\underline{x}}(t) = A_c \, \underline{x}(t) + B_c \, i_{ref}(t) \qquad (13)$$

5 - THE GUST ALLEVIATION CONTROL CONCEPTS

The blade feedback control described in the preceding section is considered as a part of a more complex longitudinal digital flight control system performing, as primary function, the helicopter stability augmentation improving its rigid dynamical behaviour and providing, when requested, the gust alleviation and associated longitudinal decoupling processes.

The G.A.C. system is conceived as a unit generating on the blade a quasi-stationary aerodynamic additional loads opposing to the gust induced airloads in order to counteract their full development; this load relaxation effect is obtained by changing the blade longitudinal cyclic pitch by a sinusoidal law:

$$\vartheta(t) = \vartheta_c \sin(\omega_f \, t) \qquad (14)$$

where the peack value ϑ_c and the frequency ω_f are propely chosen to satisfy the gust alleviation requirements. Considering (14) as the control variable for the state equation (8), it can be shown that the blade forced response will be sinusoidal with the same frequency ω_f and intensity peack which is inversely proportional to a factor $(I - r^2)$ where r is the ratio of the frequency ω_f and the first bending mode natural frequency (ω_I). Making r greater than one, the forced response will result opposite in phase to the homogeneous solution admitted by the differential equation (8). As consequence of an appropiate choice of the frequency ω_f which must be adjusted as well to yield a satisfactory matching with the helicopter dynamics disturbed by the blade flexibility effects, the the blade maximum displacement, as offered by the complete solution of Eq. 8, will be reduced. To generate additional airloads yielding an adequate counteracting effects against the gust induced airload, the maximum cyclic pitch angle has to be somehow related to the actual vibrational energy existing on the blade selected in the range of dominant natural frequencies. Furthermore the feedback augmentation effects introduced in the cyclic pitch driving signal in order to obtain the desidered artificial damping effects, must be active in the G.A.C. system process. These requirements can be simultaneusly summed up by computing, in a specified frequency slot, the spectral power density of the time function observed at the blade feedback controller and generating, by means of a function generator, a sinusoidal signal with frequency ω_f and magnitude proportional to the averaged value of computed S.P.D.; applying this signal to the input summer of the blade augmentation system, the blade will be actuated as required to implement the proposed modal strategy opposing the gust induced oscillation. In mathematical terms, referring to the Eq. (12) and defining:

$$F_b(\omega_1) = \text{D.F.T.}\left(K^T \underline{x}(t)\right) = M_b(\omega_I) + j\, I_b(\omega_I) \qquad (15)$$

the discrete Fourier transform of the feedback controller output and evaluated through a triangular (Barlett) frequency window centered at the first natural bending mode frequency (ω_I), its Spectral Power Density will be expressed by:

$$P_b(\omega_I) = M_b(\omega_I)^2 \qquad (16)$$

The maximum cyclic pitch displacement appearing in the sinusoidal law (14) will be defined as the mean value of the S.P.D. defined in (16)

$$\vartheta_c = E\left[P_b(\omega_I)\right] \qquad (17)$$

Each G.A.C. system actuation will last the time interval correspondent to a blade rotational period, i.e.:

$$T_s = \frac{2\pi}{\Omega} \qquad (18)$$

which is assumed also as reference time for in and out operations involved in the S.P.D. computations.

An other alleviation strategy aspect is related to the rotor precessional effects

induced by the cyclic pitch time variations while the blade is progressing in azimuth, which will force the rotor tip-path plane to be tilted in the longitudinal place. With a delay depending on the ratio of the aerodynamic and inertial forces acting on the blade, the helicopter attitude will vary by an amount depending upon the moments trasmitted by the blade to the hub. The vertical velocity component arising from the helicopter attitude change tends to oppose to the gust induced vertical component yielding a gust compensation effect which is particularly effective for the hingeless rotor.

Since both helicopter attitude and rotor longitudinal tilt angle are depending on the cyclic pitch changes which are proportional to the computer S.P.D. (17), the above mentioned gust compensation effect, will be proportional, although in some indirect way, to the actual vibration level existing on the blade.

If the actuating cyclic pitch actuations were in magnitude, period and phase ideally adapted to match exactly the wind gusting characteristics the helicopter should pass through the disturbance with no vertical speed perturbations. Since this condition is only theoretical, the problem to be faced is to implement a G.A.C. system not perturbing the helicopter flight path when it is operated.

For that purpose the G.A.C. system command signals are applied to the cyclic channel input of the Longitudinal Pitch Decoupling (L.P.D.) unit described in Ref. 5 the function of which is to decouple the helicopter state and control variables in two sets of properly chosen decoupled subsystems:

$$S_c \quad (d_c , \vartheta) \qquad\qquad S_{cc} = (d_{cc} , \quad w) \tag{19}$$

defined respectively the Cyclic Pitch Decoupled (C.Y.P.D.) and the Collective Pitch Decoupled (C.P.D.) channel; the first will make the helicopter attitude (ϑ) recheable only by by the cyclic pitch command (d_c) non affecting the helicopter vertical speed component and the other controlling, by means of the collective pitch command (d_{cc}) the vertical velocity component (w) independently by the longitudinal attitude. As described in Ref. 5, the L.P.D. unit is a multifeedack structure, implemented by a feedback and prefilter controllers, involving all the measurable helicopter state variables in a decoupling and regulating processes, the last one based on a specified desidered response model.

Applying the C.A.C. command signal to the (C.Y.P.D.) channel and driving the (C.P.D.) channel with a reference signal derived by a radialtimeter (R.A.) or an inertial platform (I.M.V.S.) and assigning an appropiate degree of authorithy and phasing to the cyclic and collective driving signals, the G.A.C. actuactions can be developed under the supervising control of the (C.P.D.) channel making the vertical velocity changes in respect to some reference value compatible with the tollerance predicted for the mission tollerance.

6 - THE G.A.C. SPETRAL PROCESS

The mechanization of the spectral process yielding the spectral power density informations requested in the proposed gust alleviation strategy is essential developed in two computational stages. In the first the discrete Fourier transform of the time function:

$$f(t) = K^T \underline{x} (t) \tag{19}$$

as defined in (15) is computed solving the analytical expression:

$$F_b (\omega) = \sum_{n=o}^{N-I} f (n) \quad e^{- j \frac{2}{N} k} \qquad k= O,I,..N-I \tag{20}$$

where f (n) is the time function (19) observed in time slot correspondent to the rotational blade period T_s and sampled at time intervals:

$$T_c = \frac{T_s}{N} \tag{21}$$

defining the D.F.T. computational sampling period. Since f(n) is a real, bandlimited time function, the D.F.T. spectrum (20) will a periodic symmetric conjugate frequency function defined in the frequency range:

$$F_c = \frac{\pm \pi}{T_c} \tag{22}$$

The problems arising in the spectral Power density of the function f(t) as defined in (16) are the definition, in the D.F.T. computation, of an approppiate frequency resolution al-

lowing an accurate identification of the individual spectrum component within the main lobe centered at the first blade bending mode and the evaluation of a bounded value of the S.P.D. in the range of the observed frequencies presenting narrow peacks blurred in the noise or lying on bias component. The convolution technique has been applied to face the above mentioned problems, the first of which is involved in the spectral distorsions due to truncation effects introduced in D.F.T. computation of a finite number of time function observations.

Operating initially in the time domain, the product of the Discrete Inverse Fourier Transform (I.D.F.T.) of the computed spectrum $F_b(\omega)$ by the triangular window (Barlett) discrete time function:

$$w(n) = \left(I - \frac{k}{N} \right) \qquad n = I,2,3,\dots N-I$$

is obtained and in the next computational step, the D.F.T. of this product is performed yielding a convolution in the frequency domain corresponding to the smoothed version of an infinite data sequence to an approximation which depends essentially on the window opening (N_w) at both side of the observed spectral frequency. Centering the window function $w(n)$ in the range of the dominant frequencies of interest and choosing an appropiate window width, the desidered mean value of the spectral power density as defined (17), is obtained applying the algorithm:

$$E\left[P_b(\omega_I)\right] = \frac{I}{N_w + I} \sum_{j = I - \Delta\omega}^{I + \Delta\omega} P_b(\omega_j) \qquad (24)$$

where $\Delta\omega$ is the frequency increment taken as the terminal edges of the observed frequency, the first bending mode natural frequency. The result of this computational step is the bounded S.P.D. value assumed in the G.A.C. system process as a measure of the actual energy level involved in the blade modal behaviour.

The D.F.T. of the function f(n) is computed in the present study by a F.F.T. dedicate microprocessor solving a 2-radix decimation in time - in place algorithm; this algorithm is solved by a repetitive use of a computational block (Butterfly) working on sequences of f(n) sampled applied at the butterfly input as even and odd data vectors.The computational result for the k-th spectral line, is a complex quantity:

$$F(k) = F_I(k) + W^k F_2(k) \qquad (25)$$

where $F_I(k)$ and $F_2(k)$ are the constituent spectral functions expressed by:

$$F_I(k) = \sum_{n=o}^{N/2-I} f(2nT)\, \bar{W}^{nk}$$

$$F_2(k) = \sum_{n=o}^{N/2-I} f(2nT_c + I)\, \bar{W}^{nk} \qquad n=0,I,\dots N/2-I$$

The phase factors W and $\bar{W}$ are complex vectors the numerical element of which are stored in the computer memories and repetitively employed in the F.F.T. sequential processes. In Table 4 a summary of the basic specifications regarding the F.F.T. process as it has experimented for the case treated are given.

7 - THE G.A.C. SYSTEM STRUCTURE

The constituent parts of the proposed G.A.C. system are the electro-optical Laser Sensor Unit (L.S.U.), the Spectral Processor (S.P.), the Longitudinal Pitch Decoupling (L.P.D.) unit and the Blade Feedback Controller (B.F.C.). The general topology of the G.A.C. system is given in Fig. I. The constituent units are briefly described in the following.

TABLE 4 - F.F.T. PROCESS DATA SPECIFICATIONS

Data	Dim.	Value
Measured Data max . error	N.D.	0.3%
Data Word Length	N.D.	8
Dominant Natural Frequency	Hz.	5.016
Blade Revolution Period	sec.	0.166
Observation Time Slot	sec.	0.166
Sampling Time	sec.	0.010376
Number of F.F.T. Points	N.D.	16
F.F.T. Frequency Band	Hz	96.376
F.F.T. Frequency		6
Resolution	Hz	6.0235
Clock Period	10^{-6} sec	I
Number of Butterflies Block	N.D.	32
Number of Basic operations per butterfly	N.D.	3
Time for basic operations	10^{-3} sec	0.250
Time for F.F.T. Batch	10^{-3} sec	50
Number of F.F.T. Batches for G.A.C. operation	sec	3
Number of fundamental operation per G.A.C. computational cycle	N.D.	32
Barlett window width	Hz.	
Total time per G.A.C. operation	sec	0.158

The L.S.U. Package

To measure the blade displacement in respect to a reference datum and its velocity described respectively by the state variables g(t)and $\dot{g}$ (t) in the state equation (8), an electro-optical Laser sensor, which is a particualar application of the laser position encoder employed for shape measurements in the large structure in space (Ref. 6,7), is proposed for the G.A.C. system implementation. This device uses a coaxial trasmitter-receiver pulsed diode laser with a swepting capability in a specified angular range. By means of a photo-sensitive detector, the angle of the laser beam streaking a reflector point fixed on the blade and the distance of the reflector from the laser collimator lens, are measured. The basic principle in this measurement process is that the emitted returned returned radiations from the reflector detected by the laser head are imaged, with time delay introduced by a fiber optic unit, on a phosensitive array and the coordinates of the imaged spot are observed and measured. The signal at the photosensitive L.S.U. output is processed by a microprocessor where the geometric relation between the two measured quantities is solved for the discrete displacement function g(n) of the observed reflector point in respect to a reference datum. If a number of reflectors located at different points along the blade are included in the

design and utilizing the L.S.U. scanning capability, particular g(n) functions and the twist angle relative to the different blade sections may also be computed. The L.S.U. microprocessor is employed as well to evaluate the incremental changes in the discrete computed function g(n), yielding, with an approximation compatible with the G.A.C. process resolution, the discrete function $\dot{g}$ (t).

Applying the single blade control concept, the laser head, which is solidal to the helicopter fixed frame, will be directed to observe a reflector point located in a position along the blade, 0.7 R in the present application, which depends from the rotor configuration.

The basic augmented helicopter flight control system

The basic longitudinal flight control system for the helicopter taken into consideration in Ref. 5 consists of a multi-feedback structure involving all the helicopter state variables which are regulated, in order to obtain the desidered dynamical behaviour, by means of a set of feedback gains. The governing equations for the longitudinal autopilot - vehicle system will be:

$$\dot{\underline{x}}(t) = A_H \, \underline{x}(t) + B_H \, \underline{u}(t) \tag{28}$$

$$\underline{u}(t) = K_h \left(\underline{r}(t) + K_{fh} \, \underline{x}(t) \right) \tag{29}$$

where the state and control vectors are defined:

$$\underline{x}(t) = \left(u(t),\ w(t),\ q(t),\ \vartheta(t) \right) \tag{30}$$

$$u(t) = \left(d_c(t),\ d_{cc}(t) \right) \tag{31}$$

The forward and feedback matrices K_h and K_{fh} are computed assuming as the primary variables to be controlled by the cyclic and collective channels, respectively the helicopter attitude (ϑ) and the vertical velocity component (w) as indicated by the output equations:

$$y_c(t) = C_c \, \underline{x}(t) = \vartheta(t) \tag{32}$$

$$y_{cc}(t) = C_{cc} \, \underline{x}(t) = w(t) \tag{33}$$

The feedback and forward gains for the two autopilot channels are given in Table 4.

TABLE 4 - AUTOPILOT FEEDBACK GAINS FOR THE AUGMENTED AUTOPILOT

V = 56.5 m/sec, S.L.

Gain	Sensor	Cyclic Channel	Collective Channel
K_u	I.M.V.S.	- 0.35133	- 0.030158
K_w	"	- I.32133	- 2.38311
K_q	R.G. "	- 11,5025	- 20.6326
K_ϑ	V.G.	0.227311	0.40774

The helicopter dynamical behaviour improvements obtained by the autopilot feedback closure are clearly indicated by the closed loop roots location in the complex plane resulting from the characteristic equation:

$$D(s) = (s + \lambda_I)\ (s + \bar{\lambda}_I)\ (s + \lambda_2)\ (s + \bar{\lambda}_2)$$

TABLE 5 - HELICOPTER DYNAMICS (V = 56.5 m/sec, S.L.

	Bare Helicopter	Augmented Hilicopter
Short period (λ_I)	- 0.958 $\pm$ j 0.2548	- I.0737 $\pm$ j I.858
Phugoid (λ_2)	0.0C44 $\pm$ j 0.415	- 0.0116 $\pm$ j 0.104

The L/P.D. unit

The state and control variables involved in the augmented helicopter are transformed in two sets of decoupled subsystems (19) which are forced to respond in similarity to a specified response model by means of a regulating process which, together with the decoupling process, is implemented by the control law:

$$u(t) = K' \underline{x}(t) + G' v(t) \tag{34}$$

where K' and G' are respectively the feedback and forward matrices (Fig. 4) computed applying the generalized computer program described in Ref. 9 and requiring among the other basic input data, the information relative to the desidered dynamics established for each of the two decoupled subsystems (19). Since the decoupling process is expected to be employed to support the gust alleviation process and being essential in such event that the helicopter, controlled through the L.P.D. unit has the same dynamics predicted for the augmented halicopter, an unique response model must be adopted for the two cases; in this way the same helicopter behaviour will be experienced during the G.A.C. system operations in which the feedback closures implemented by the gains indicated in Table 5 will be substituted by the L.D.A. controller gains the value of which, with the prefilter gains, are indicated in Table 6.

TABLE 6 - L.P.D. FEEDBACK AND PREFILTER GAINS
V = 56.5 m/sec; S.L.

Gain	Variable	Sensor	Subsystem S_c Cyclic channel	Subsystem S_{cc} Coll. Channel
$\hat{g}_{II}$	d_c	Trx	- 0.139381	
$\hat{g}_{12}$	d_{cc}	"	- I.41754 IO^{-3}	
$\hat{g}_{21}$	d_c	"	-	- 0.02711
$\hat{g}_{22}$	d_{cc}	"	-	- 0.01155
K_{II}	u	I.M.V.S.	2.8923 10^{-3}	
K_{12}	w	"	- I.9I457 10^{-3}	
K_{13}	q	R.G.		
K_{14}	ϑ	V.G.	-	- I.568I 10^{-4}
K_{21}	u		-	- I.568I IO^{-4}
K_{22}	w	"	-	- 9.39256 10^{-4}
K_{23}	q	R.G.	-	0.625952
K_{24}	ϑ	V.G.	-	-

8 - THE G.A.C. SYSTEM PERFORMANCES EVALUATION

The effects of the G.A.C. system performances on the rotor dynamics, when it is disturbed by a random disturbances simulating severe gusting conditions, were evaluated in an extensive computer simulations employing the mathematical models given in the preceding sections. A stochastic model was employed to simulate the gust disturbance which has been considered as a forcing function applied to the rotor cyclic pitch control in a time interval correspondent to 90° "master" blade rotation within the azimuth sector (45° 135°) in which the G.A.C. "identification" stage is carried out; in this time interval, the function g(t) is observed in 16 discrete points and its spectral power density is computed making it available for the next G.A.C. system actuation stage. Here the sinusoidal forcing function, modulating the blade in pitch, is generated in the attempt to attenuate the vibrational energy level measured in the preceding identification stage. The gust effects are updated at the beginning of the next identification stage in which a new S.P.D. value is computed and the process will continue through the successive blade revolutions. If the gust function is isolated and concentrated in the identification spatial sector, and the blade dynamic is damped, the computed S.P.D. will be progressi-

vely reduced to reach a minimum observable value and, at this time, the forcing function is automatically excluded and the G.A.C. system becomes inactive until the effects of a new gust function appear. As it was considered in the preceding section, the resulting rotor dynamics are strongly dependent on the degree of the artificial damping introduced in the augmented blade and on the frequency of the forcing function actuating the active process. Several computer runs indicate that the best ratio of the forcing function to the first bending mode natural frequency, is, for the case treated, ranging from I.5 to 2 (value assumed I.9) and that the number of blade revolutions required to reduce the computer S.P.D. value from its initial value, strictly dependent on the gust intensity level, to a specified limiting bias value is dependent on the equivalent time constant relative to the augmented blade. The number of the revolution required to reach the established S.P.D. limiting value and the mean value or the integral squared value of the function g(t) observed during the G.A.C. actuation stage, are the performance indexes assumed, for comparison purpose, in the G.A.C. system performances evaluation. To investigate the effects of the rotor flexibility on the helicopter dynamical behaviour, the moment differential equation relative to the helicopter short period mode mathematical model was augmented with the terms relative to the center of gravity moment trasmitted by the gust disturbed oscillating blade to the helicopter rigid frame through the rotor hub. The same gust excitation technique used for the rotor, was applied for the helicopter-flexible rotor dynamics simulation with the only difference that the cyclic pitch input commands were applied to the longitudinal pitch decoupling unit. After initial computing trials, directed to check the L.P.D. unit effectiveness in the decoupling and regulating the helicopter cyclic and collective channels, the results have confirmed the positive conclusions reached in the previous research study (Ref. 5) allowing the variable w(t) to be omitted from the approximate mathematical model for which only the cyclic pitch control is considered.

In the simulation the blade displacements are excited, during the G.A.C. system identification stage, by a random function with a standard deviation of I.4 m/sec and the correspondent record is displayed in Fig. 6. The blade energy level recorded at the spectral processor output results 0.354 N.m.; in the following actuaction stage this value was assumed as the amplitude of the sinusoidal forcing function, having a frequency of 60 rad/sec., modulating the blade cyclic pitch for the time interval in which the blade is completing its first revolution. At the end of the next identification stage, in which only the blade residual vibrational motion resulting from the preceding actuation stage, was observed by the L.S.U., the updated blade energy level was 19% lower than the preceding value and it continues to decrease, see Fig. 7, in the third blade revolution after that it becomes practically a costant with a value below the sensitivity limit, 10^{-3} N.m., proposed in G.A.C. system operations; this terminal blade energy level represent essentially an high frequency, low amplitude vibration which was removed taking the sinusoidal forcing function out from the cyclic pitch channel actuation. In the next simulation step the rigid-helicopter-flexible blade model was taken into consideration maintaining the same gust disturbance and the results indicate that, after an initial gust induced helicopter attitude variation, it begins to reduce exponentially falling rapidly to zero as the blade energy level has reached the observed constant value, indicating that the residual blade vibration is not influencing the helicopter transient behaviour which is resolved in a time interval correspondent to the helicopter short period mode time constant.

9 - CONCLUSIONS

The objective proposed in the present research was essentially to evaluate in the most simple and explanatory way the gust alleviation technique based on the spectral informations obtained by a F.F.T. dedicate microprocessor working on the vibrational data observed by an electro-optical sensor. The presented simulation results, among others not presented for reason of work length, indicate the G.A.C. system feasibility and applicability particularly for combat helicopters which must operate in severe environmental condition which may be experienced particularly in high speed, low altitude penetration missions where rigid tollerance limits are imposed on the flight path profiles.

AKNOWLEDGMENT

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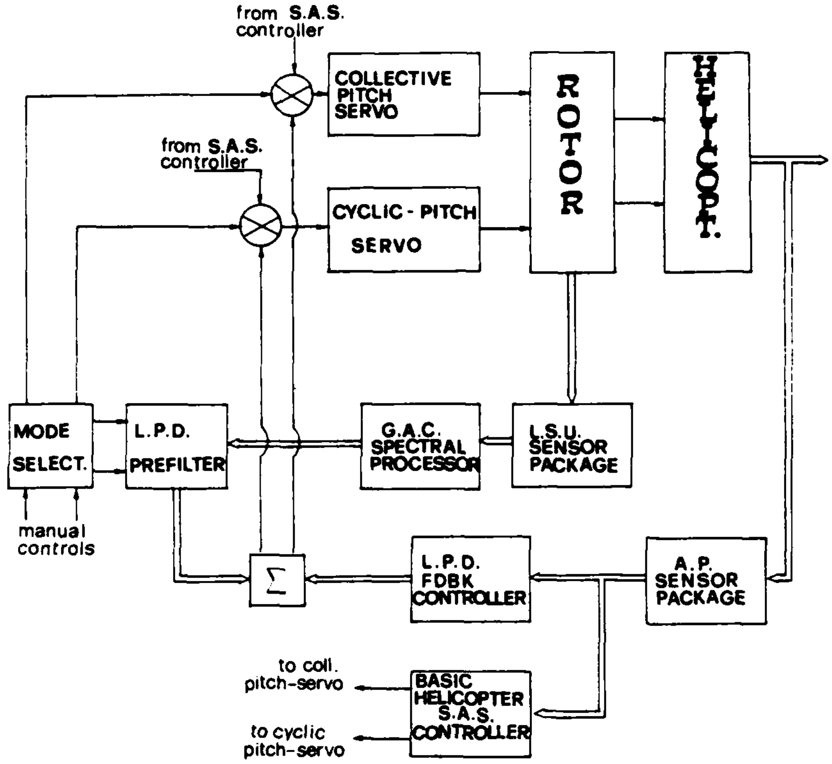
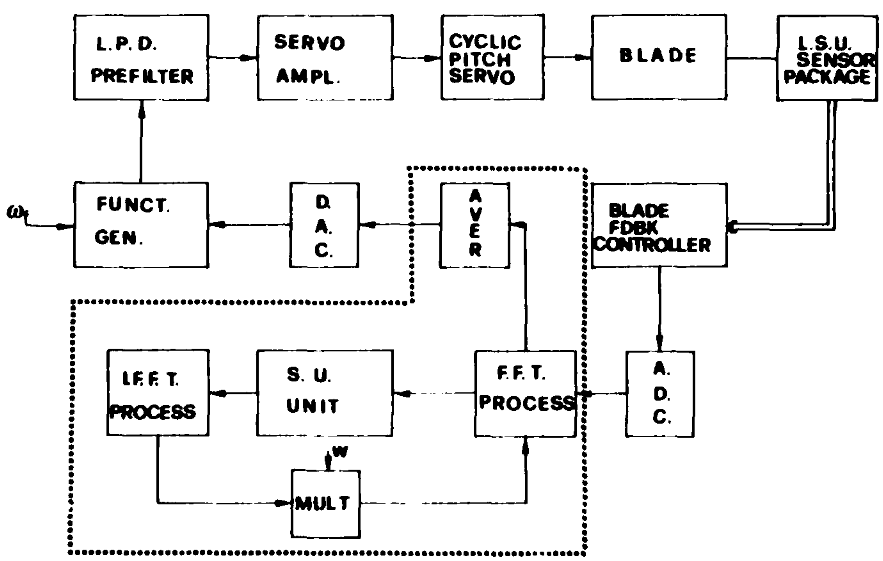


FIG. 1



G.A.C. SPECTRAL PROCESSOR

FIG. 2

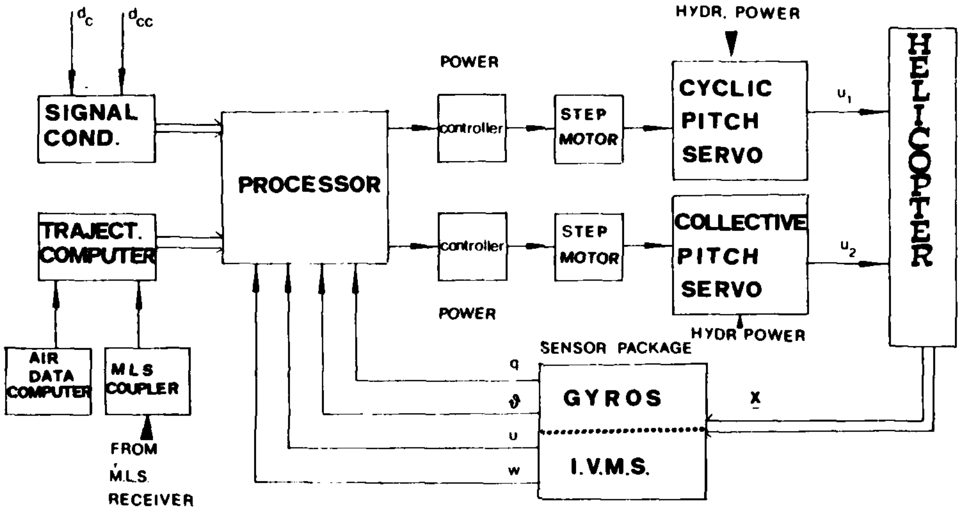
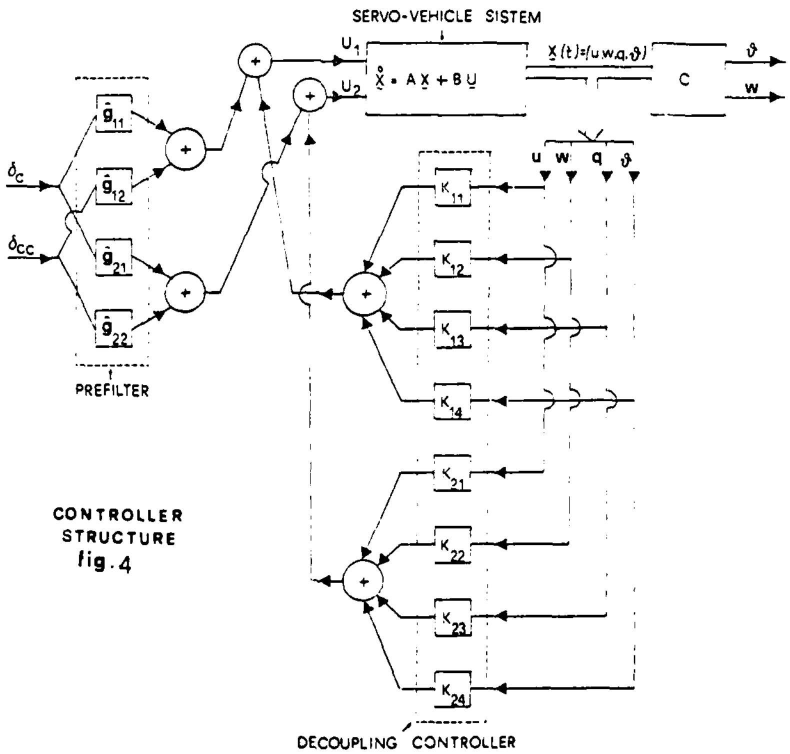
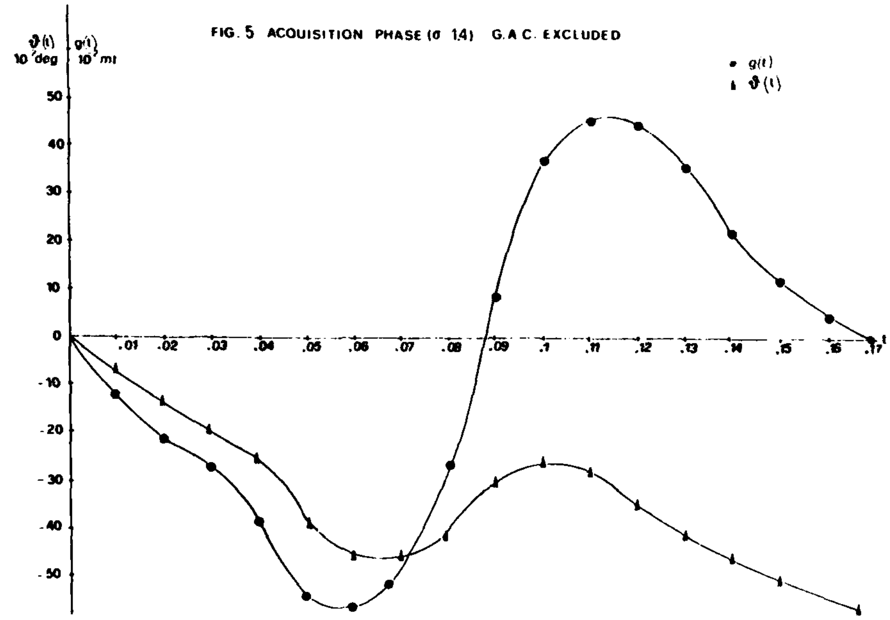


FIG. 3. FLIGHT CONTROL SYSTEM IMPLEMENTATION



CONTROLLER STRUCTURE
fig. 4

FIG. 5 ACQUISITION PHASE (σ 1.4) G.A.C. EXCLUDED



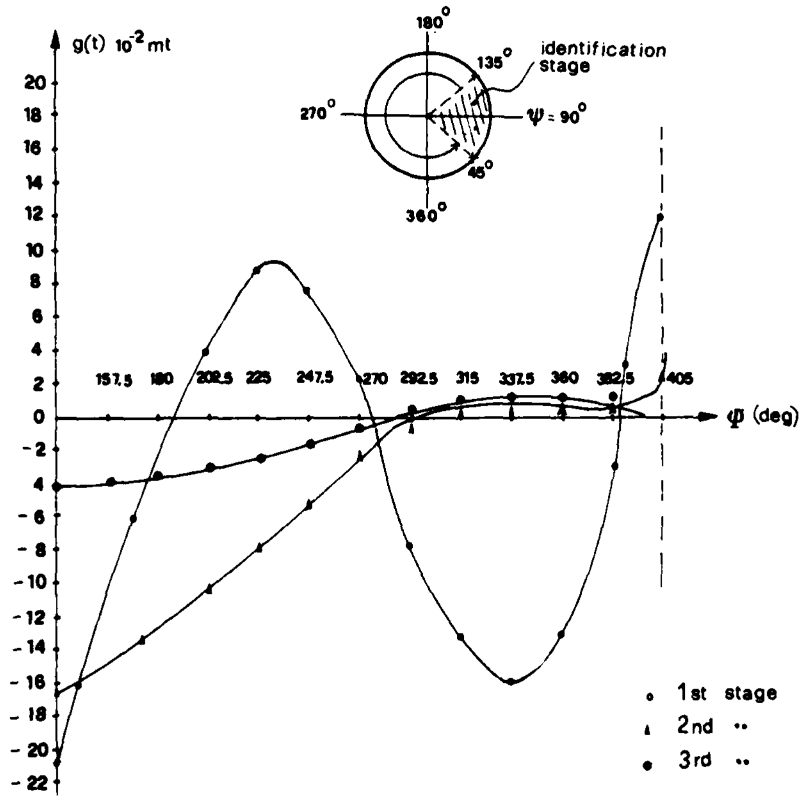


FIG. 6 - L.S.U. OBSERVED g(t) FUNCTIONS

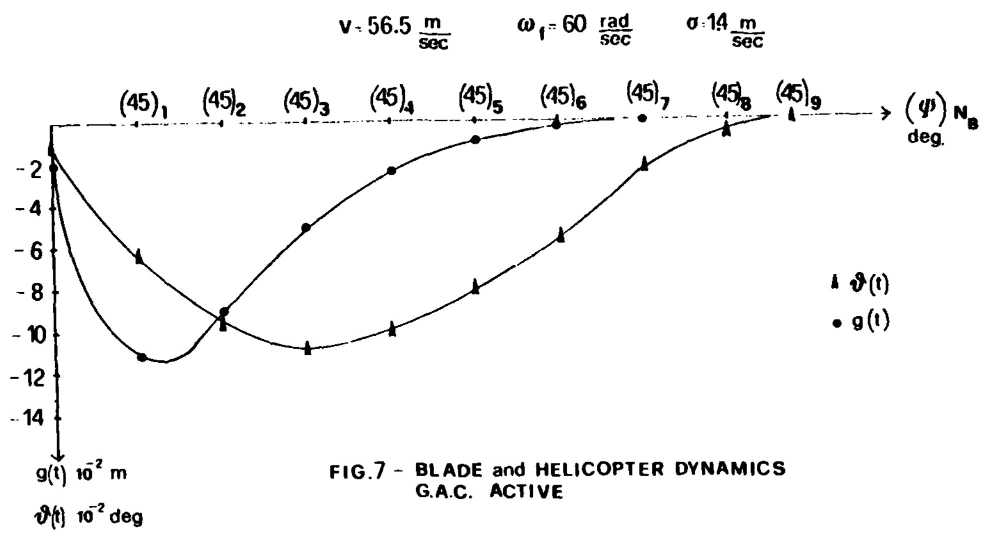


FIG.7 - BLADE and HELICOPTER DYNAMICS
G.A.C. ACTIVE

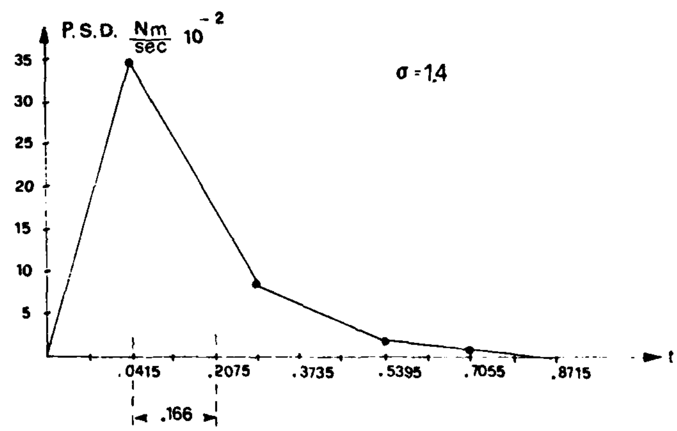


FIG. 8 - P.S.D. PROCESSOR OUTPUT

FLIGHT RESEARCH ON VISUAL AIDS AND NAVIGATION EQUIPMENT FOR HELICOPTER LOW-LEVEL FLIGHT AT NIGHT

by

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SUMMARY

An avionic/optronic system comprising night goggles, electronic head-down display and doppler navigation was tested in a Bo 105 helicopter in low-level flight at night. Particular emphasis was given to the assessment of system performance and pilot workload. The results obtained as well as the methods and procedures applied are discussed in a way that similar experiments in the future may benefit from the outcome.

1. INTRODUCTION

Currently two major helicopter projects are sponsored by the German Government: The tank attack helicopter PAH2 (Panzerabwehrhubschrauber 2) and the light transport helicopter/search and rescue (LTH/SAR). While the PAH2 is a pure military project, the LTH/SAR also offers a lot of promising features for civil applications. It is not unrealistic, therefore, to expect a joint military and civil effort for at least some subsystems of the LTH/SAR.

Common to both military and civil applications is the requirement that the helicopter can be operated at night. DFVLR, therefore, initiated an appropriate research program in the early seventies and flight tested a range of suitable equipment like FLIR, LLTV, night goggles, HMS/D and electronic head-down displays. The research is coordinated with related industrial programs by the German Department of Defense.

From the experiences gained from these equipment tests and from discussions with the military it was concluded, that a system comprising night goggles, electronic head-down display and doppler navigation would serve the low-level flight of helicopters at night and would have certain advantages. These included low complexity, moderate cost and easy installation and maintenance. However, system performance and resulting pilot workload had to be determined.

2. PURPOSE OF RESEARCH

Figure 1 presents a typical flight profile for a battlefield scenario expected in Middle Europe /1/. Progressing towards the front line of the battlefield the helicopter is required first to fly about 40 % of the mission below 300 ft agl (region D), then another 40 % below 150 ft agl (region C), followed by 10 % of the mission to be flown below 100 ft agl (region B) and finally to fly the last 10 % of the mission around 10 ft agl (region A).

One purpose of the research was to gather experience on the performance of an avionic/optronic system, suggested for the low-level flight at night, in regions B and C. These regions represent about 50 % of a transport mission in the scenario depicted in fig. 1 and require a maximum altitude of 150 ft agl to be maintained.

Another purpose of the research was to assess pilot workload. Prior to the experiments it was anticipated that pilot workload would reach a high and temporarily unacceptable level. Possible causes are the combined effects of limited vision, risk of collision with obstacles, imbalanced weight of the helmet-mounted night goggles, working in a dark cockpit, frequent navigation updating, lack of radio navigation aids, susceptibility of the navigation system to variations of the earth magnetic field and to the magnetic field of power lines, unfamiliar aspect ratio of the terrain, heat and noise, necessary frequent communication among crew members, excitement, perspiration and anxiety - to name a few.

The more specific questions to be answered by means of the flight research were:

- How well does the particular avionic/optronic system assist the crew in performing the mission?
- Is pilot workload kept at an acceptable level?
- How does the level of ambient illumination affect flight performance?
- Is the electronic head-down display of instrument information accepted by the pilots?

The purpose of research, the design of the experiment and the results obtained are published in greater detail in /2/.

3. DESIGN OF THE EXPERIMENT

The avionic/optronic system to be flight tested was composed of

- night goggles (Electro Special, Type BM 8028)
- electronic head-down display (DFVLR)
- doppler navigation system (Standard Elektrik Lorenz AG, Type AN/ASN 128 (XE-2)

The tests were flown in the Bo 105 - S123 helicopter of DFVLR. Figure 2 shows the test pilot wearing night goggles, the electronic head-down display and the control panel for the doppler navigation system. The night goggles were mounted on the helmet at some distance from the eyes so that the pilot could monitor the electronic head-down display directly.

Figure 3 presents the details of the head-down display symbology. FIRE, CHIP, HYD1, HYD2, H-BL, GEN1, GEN2, T-OL, K-OL, BAT and FILT warnings were displayed in the hatched areas of fig. 3. Two blinking diagonal lines crossing the whole display area served as master warning indicator. Identical auditory warnings were generated simultaneously by a speech synthesizer.

Three pilots formed different test crews who flew a total of 70 low-level transport missions. Most of the test flights took place at night but some of them were conducted under ordinary daylight conditions with standard cockpit instruments for reference purposes.

It was expected in general that the pilots would fly at a flight level well below 150 ft agl. Rather than commanding a specific maximum upper flight level the pilots were instructed to

- fly as low as possible
- maintain an airspeed of at least 80 kt
- follow as accurately as possible a given track

According to previous flight experiences, duties were assigned to the two crew members as follows:

- The navigator (right seat) was responsible for pilot supervision and navigation. He informed the pilot about necessary changes of the track, peculiarities of the terrain and obstacles to be expected using a map scaled 1:50000. He also operated the doppler navigation system.
- The pilot was responsible for flying the helicopter safely. He maintained the demanded track and reported recognized and identified obstacles to the navigator. Furthermore he had to acknowledge a specific TEST indicator, showing up on the head-down display irregularly, by pressing a switch on the control stick.

The data recorded in the experiments included

- technical parameters like airspeed, radio altitude, stick motion etc.
- behavioural parameters like response time, eye-point-of-regard etc.
- physiological and biochemical parameters like ECG, EOG and urinary excretion of hormones
- pilot's comments gathered by means of rating scales (Figure 4) and a structured questionnaire.

The experimental design and the analysis of data followed the guidelines for psychological research as laid down, for example, in /3/.

4. RESULTS

The aims of the research could be achieved largely by the outcome of the flight tests and the subsequent analysis of the data. In the light of the specific questions formulated earlier the following results were obtained.

a) Avionic/optronic system performance

The pilots considered the system - night goggles, electronic head-down display and doppler navigation - adequate to perform a low-level transport mission at night. For an ambient illumination down to 5 millilux the pilots were able to maintain a flight level of less than 100 ft agl at an airspeed of more than 80 kt. For lower levels of ambient illumination and for flights over unfamiliar terrain greater average flight levels of 104 resp. 127 ft agl accrued. However, these were well below the 150 ft agl criterion set prior to the tests.

Accurate navigation was achieved by a combination of terrestrial and doppler navigation. Pilot comments showed no statistically significant differences between low-level flights at night and under daylight conditions regarding maintenance of the track, monitoring the terrain, obstacle clearance, monitoring the instrument displays, flight performance and experienced workload.

However, the "adequate" rating of system performance by the pilots does not imply "comfortable". Several improvements were considered desirable including stability augmentation (not available on DFVLR Bo 105 helicopters so far), immunity of the magnetic heading reference sensor of the doppler navigation system to the electro-magnetic field of power lines and a special cockpit lay-out for the night vision environment.

b) Pilot workload

The demands on the pilot were significant, and table 1 presents as an example the number of necessary conversations between crew members. During a flight period of 25 minutes, about 180 pieces of information were exchanged which corresponds to 1 information exchange every 8 seconds. Therefore, the necessary conversation among crew members occupied a significant portion of their available time budget. The analysis of response times to the TEST indicator in the electronic head-down display supports this view. This indicator was not connected to the aural warning system and could be detected only visually on the display. While in ordinary daylight flights the median of response times was 3.1 seconds, it became 8.6 seconds for low-level flights at night. With a range of about +/- 60 % around the median, maximum response times on the order of 13-14 seconds were measured occasionally. This, again, signals a high time-load on the pilots. In contrast, the warning indicators showed-up on the electronic head-down display and on the aural warning system simultaneously. Consequently, the response times to these signals were accordingly low.

It was surprising, therefore, that the pilots did not rate workload significantly higher for low-level flights at night than for identical flights under daylight conditions. The physiological/biochemical parameters obtained from one pilot support this view. Average pulse rate was surprisingly low (75-88 bpm) with maximal values between 80 and 95 bpm. Furthermore, figure 5 presents as a reference the circadian rhythms in the normal urinary excretion pattern of hormones of a group of 8 subjects over six 24-hour periods /4/. The average excretion rates of adrenaline (2.4 μg/3 h), noradrenaline (4.8 μg/3 h) and 17-OHCS (35 μg/3 h) measured during the low-level flights at night are well within the ordinary circadian variation of these parameters. The excretion rates measured at night during the flights also correspond well with rates normally expected for daylight hours. This suggests that pilot activation during the flights was comparable to ordinary working hours. All together, there was no sign of extraordinary workload.

c) The effects of low ambient illumination

It was anticipated that each generation of night goggles enables the pilot to maintain daylight flying strategy until ambient illumination drops below a certain level. No major change of flying strategy compared to daylight flights was experienced down to a level of ambient illumination on the order of 5 millilux. However, for an ambient illumination of less than about 5 millilux

- the variance of stick movements was reduced by 15 % in pitch and by 30 % in roll

- the activity of the helicopter in turn, described by

$$A = \frac{\Theta}{2} (\omega_{z,rms})^2$$

Θ - moment of inertia around the vertical axis

ω_z - angular velocity around the vertical axis

  was reduced by about 45 %

- the amount of non-coordinated flying with respect to bank and turn, described by the squared product-moment coefficient of correlation of bank angle and turn rate, was reduced by about 55 %.

All together, these findings demonstrate that for an ambient illumination below a certain level - in this case 5 millilux - the pilots changed their flying strategy from taking full advantage of the handling qualities of the helicopter to a more cautious handling of the vehicle. This goes along with a loss of mission performance as the helicopter is flown less well adapted to the peculiarities of the terrain. And this, in turn, is considered an unambiguous sign that the operational limits of the system - in particular those of the night goggles - were approached.

d) Acceptance of the electronic head-down display

While the night goggles and the doppler navigation system were only additions to the helicopter, the electronic head-down display was a critical replacement of the existing conventional instruments. But the pilots rated the readability of the elec-

tronic head-down display significantly ($p < 0.001$) better than that of the conventional instruments, and they appreciated the integration of all vital indicators in one centralized display. They unanimously voted that almost too much information is displayed simultaneously. Two pilots suggested different modes to unburden the display, i.e. a differentiation between the symbology required for low-level flight vs. the symbology required for IFR flight, because it is unlikely that both modes are required at the same time. Furthermore the pilots reported that the various warning indicators were easy to identify and they rated their combination with identical aural warnings very efficient.

Eye-point-of-regard measurements showed that the pilot's vertical angle of view - referenced to the horizon - covered a range of approximately 0 degree (search for navigational fixes) to -15 degrees (passing an obstacle). The latter value became -30 degrees sometimes in cases of very low visibility. According to these results it was concluded that a head-down display in front of the pilot should clear a vertical field of vision on the order of -30 degrees and should fit within a vertical field of vision between -30 degrees and -50 degrees. This constitutes a compromise between unrestricted observation of the terrain and comfortable monitoring of the head-down display. If greater emphasis is placed on pilot's unrestricted view, the display may be moved to a side or center panel position.

e) General aspects

The purpose of the research, i.e. the assessment of a specific system for helicopter low-level flight at night, has been largely achieved. The results were appreciated by those involved in the definition, design and operation of advanced helicopters. But the experiences gained from this research also emphasize some more general aspects.

One aspect is the employment of a marked change of pilot's flight strategy as a criterion to determine the operational limits of a given system. In the experiment ambient illumination below about 5 millilux was the fundamental cause for such a change. Improved night goggles may shift the occurance of this change to a lower level of ambient illumination. But the introduction of command/stability augmentation which unburdens the pilot in a different way might have led to a similar effect. However, the criterion would reflect the combined effects of both subsystems on flight performance. Therefore, the criterion of a change of flight strategy is considered operationally relevant and efficient to explore the overall limits of system performance under changing environmental conditions.

Some time ago an AGARD Working Group listed in a [illegible]y 55 more or less different parameters to measure workload in aviation. In the mean-time this number may have more than doubled. But experience shows that only few parameters have a chance to be generally accepted. In the experiment, therefore, only those methods and parameters were utilized which

- are well-known physical quantities (time, power, bank angle, eye-point-of-regard etc.) in contrast to, for example, a quantity described as "eye blinks per bit of information displayed"

- have found acceptance in different fields of research, e.g. subjective ratings, biochemical analyses, behavioural data etc.

- are accepted standards, e.g. statistical methods

The major advantages are, that such methods and parameters are almost generally available and that at least in one field of research some form of a standard exists. The importance of adrenaline / noradrenaline as stress-indicators, for example, is well known and a lot of experience exists regarding the relationship between these parameters, known stressors (acceleration, extended work periods, rapid change of time zones etc.) and resulting human performance and reliability. The same applies to the known effects of heavy time-load, to the design of unbiased subjective rating scales etc. Therefore, the set of parameters chosen for the experiment is considered balanced with regard to the many facets of workload. However, possible improvements are being investigated for future experiments including the subjective workload analysis technique (SWAT) as well as voice stress analysis (VSA).

5. FUTURE PLANS

The pilots expressed their desire to continue the test flights and to get more experience with low-level flights at night over unfamiliar terrain at different speeds and with landings of varying difficulty. Even more, they suggested an identical test programme for the next (third) generation of night goggles and that a greater number of pilots should participate in the experiments.

It was felt, however, that in such a programme more emphasis would be placed on pilot training with existing systems for low-level flight at night than on the development of more advanced systems. DFVLR, therefore, decided to give greater priority to the research on command/stability augmentation systems, fly-by-wire/light technology, digital flight control, new types of displays and controls and improved sensor and navigation systems in the future.

DFVLR's new HETAS (Hubschrauber Erprobungs-Träger für Avionik-Systeme) is an experimental system comprising two Bo 105 helicopters and corresponding simulation facilities on the ground serving this future programme. While the Bo 105-S3 helicopter will carry a triple-redundant fly-by-wire/light control system, the Bo 105-S123 helicopter is employed as a test vehicle for other avionic and electro-optical systems. Both lines of flight research may eventually merge and be transferred to a BK117 helicopter serving both as a test vehicle and technology demonstrator for integrated helicopter avionic systems.

The HETAS facilities on the ground are centered around a VAX 750 computer which controls a helicopter fixed cockpit flight simulation and supports software generation, hardware test and integration of helicopter avionic systems. The airborne and ground facilities of HETAS will have compatible system structures and subsystem interfaces so that experiments successfully completed on the ground can be easily transferred to the helicopters for flight tests.

HETAS will be available to both industry and DFVLR to evaluate and demonstrate avionic subsystems and their integration into helicopter avionic systems. DFVLR provides the methods and procedures necessary for test, evaluation and documentation.

6. CONCLUSIONS

From the results of the flight research the following conclusions were drawn:

The performance of the investigated system - comprising night goggles, electronic head-down display and doppler navigation - was considered adequate for a low-level transport mission at night. However, performance limits were approached for very low levels of ambient illumination and for flights over unfamiliar terrain. These limits may be overcome to a large extent if better visual aids become available.

The mission to be flown was not considered particularly difficult for trained pilots. Pilot workload was found to be at an acceptable level.

The readability of the electronic head-down display was better than that of the conventional instruments. The integration of all vital instruments in the display was a significant factor contributing to the good overall performance of the system.

An unobstructed view of the terrain in the vertical plane down to at least -30 degrees with reference to the horizon is recommended. An electronic head-down display in front of the pilot can be comfortably monitored below this area at a vertical angle of view between -30 and -50 degrees.

Pilot's flight strategy, characterized by pilot control action, helicopter motion and the degree of coordinated flight with respect to bank and turn, mainly depends on the difficulty of the task, on the performance of the technical system and on the safety level assumed by the pilot. For a constant task, a noted change of control strategy - as experienced in the flight tests - therefore is considered an indication that system performance limits are approached.

The employment of generally available, non-exotic methods to assess workload produced results which had a large degree of objectivity and mutual agreement. The results are easy to interpret by those involved in the definition, design and operation of helicopters.

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8. ACKNOWLEDGEMENTS

The paper is based largely on research by Messrs. E. Danneberg, E. Kohnen and Dr.rer.nat. F.V. Schick (DFVLR, Institute for Flight Guidance), Dr.phil. R. Hann (WPAFB, Aerospace Medical Research Laboratory) and Dr.med. H.M. Wegmann (DFVLR, Institute for Aviation Medicine). The test flights were performed in close cooperation with the Army Aviation School of the German Army Aviation.

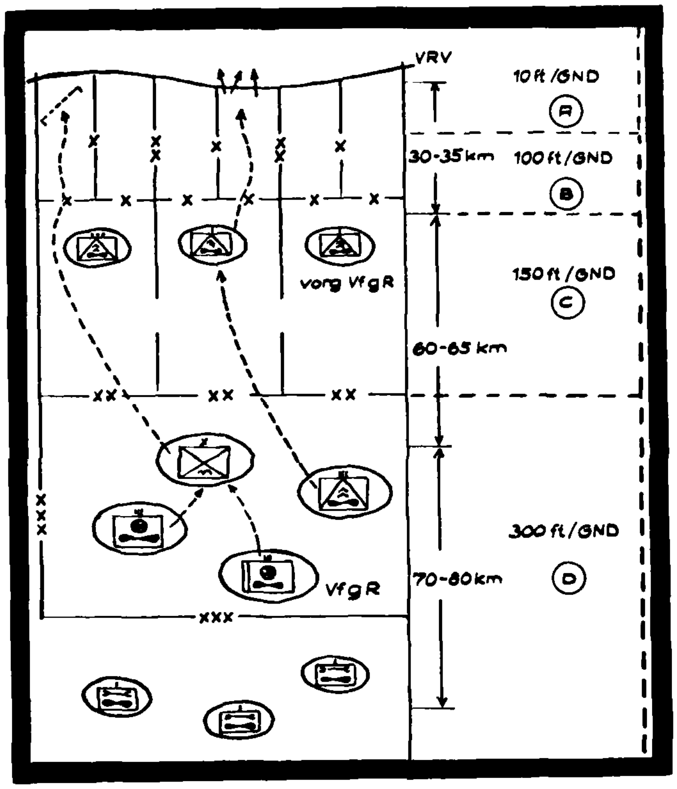

Fig. 1 Battlefield Scenario expected in Middle Europe /1/. Maximum altitudes not to be exceeded in flight are identified by regions A-D.

Fig. 2 Bo 105 helicopter. The night goggles are mounted on the helmet at some distance from the pilot's eyes so that he can monitor the electronic display in front of him directly. The controls for the doppler navigation system are under the pilot's right knee.

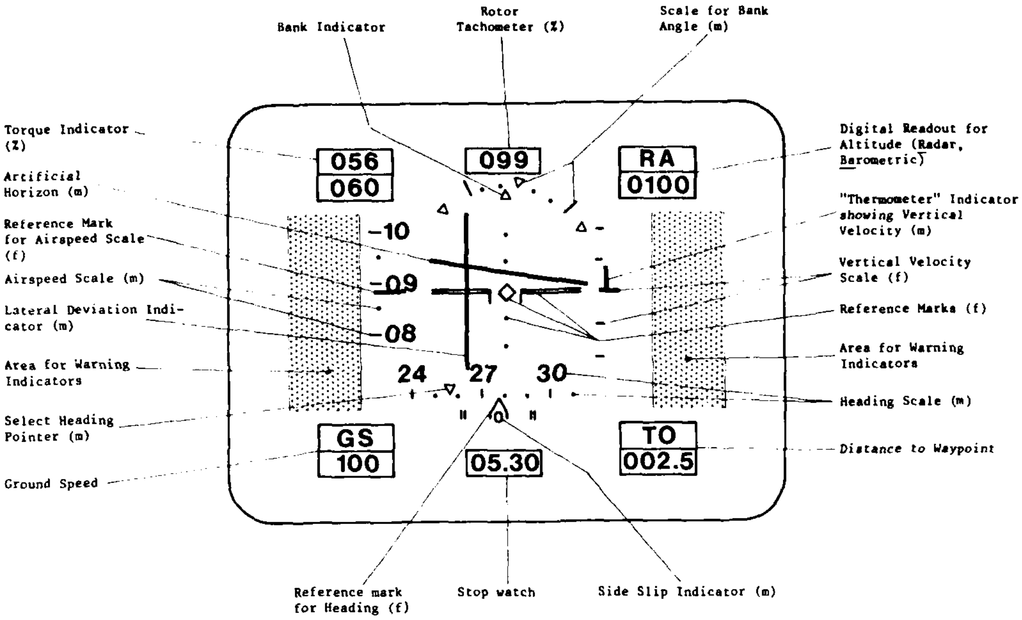

Fig. 3 Lay-out of the electronic head-down display.

	COMPLETELY TRUE	MOSTLY TRUE	SOMEWHAT TRUE	LESS TRUE	BARELY TRUE	ABSOLUTELY NOT TRUE
IT WAS DIFFICULT TO KEEP THE TRACK EXACTLY	○	○	○	○	○	○
I HAD ENOUGH TIME TO OBSERVE THE TERRAIN	○	○	○	○	○	○
I CAME CLOSER TO OB- STACLES THAN I WANTED	○	○	○	○	○	○
I DID NOT HAVE ENOUGH TIME TO LOOK AT THE INSTRUMENTS	○	○	○	○	○	○
I COULD FLY AS WELL AS I DO IN DAYLIGHT	○	○	○	○	○	○
I WAS VERY TENSE	○	○	○	○	○	○
I COULD EASILY READ THE NECESSARY INSTRU- MENTS	○	○	○	○	○	○

Fig. 4 Pilot's rating scales.

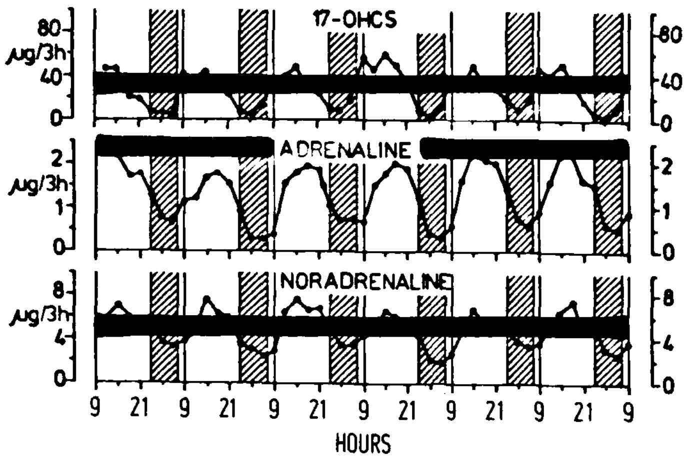

Fig. 5 Urinary excretion levels of hormones during low-level helicopter flight at night (1 pilot). The underlying graph of the circadian rhythm in the urinary excretion pattern of the hormones (from /4/) is presented as reference. The hatched areas represent night periods.

NAVIGATOR	FLIGHT A	FLIGHT B
NAVIGATION ANNOUNCEMENTS	46	47
NAVIGATION CONFIRMATIONS	7	17
SPECIFIC ALTITUDE ANNOUNCEMENTS	1o	5
SPECIFIC OBSTACLE ANNOUNCEMENTS	31	25
QUERIES AND OTHER ANNOUNCEMENTS	14	11
TOTAL	1o8	1o5
PILOT		
NAVIGATION CONFIRMATIONS	2o	35
NAVIGATION QUERIES	15	11
ALTITUDE CONFIRMATIONS	5	5
OBSTACLE CONFIRMATIONS	28	13
ALL OTHER ANNOUNCEMENTS	5	7
TOTAL	73	71
GRAND TOTAL	181	176

Table 1 Number of verbal information exchanges of navigator and pilot during low-level helicopter flight at night. The results of two different flights are presented for comparison.

DEVELOPMENT AND FLIGHT TEST OF A HELICOPTER COMPACT, PORTABLE, PRECISION LANDING SYSTEM CONCEPT

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SUMMARY

A novel, airborne, radar-based, precision approach concept is being developed and flight tested as a part of NASA's Rotorcraft All-Weather Operations Research Program. A transponder-based beacon landing system (BLS) applying state-of-the-art X-band radar technology and digital processing techniques, has been built and is being flight tested to demonstrate the concept feasibility. The BLS airborne hardware consists of an add-on microprocessor, installed in conjunction with the aircraft weather/mapping radar, which analyzes the radar beacon receiver returns and determines range, localizer deviation, and glide-slope deviation. The ground station is an inexpensive, portable unit which can be quickly deployed at a landing site. Results from the flight test program show that the BLS concept has a significant potential for providing rotorcraft with low-cost, precision instrument approach capability in remote areas.

INTRODUCTION

A self-contained navigation system requiring minimum ground-based equipment is desirable to make full use of the helicopter's unique capability of remote-site, off-airport landings. In pursuing this goal, the NASA Ames Research Center is conducting cooperative research with the University of Nevada, Reno, and the Sierra Nevada Corporation to develop the use of airborne weather radar as a primary navigation aid for helicopter approach and landings in instrument flight rules (IFR) conditions. In the first phase of this effort, the detection of passive ground-based corner reflectors using a device called an echo processor was successfully demonstrated (Ref. 1). Use of this passive-reflector detection scheme in the overland environment provides pilots with a target on their radar display, and gives them the range and bearing information necessary for a nonprecision approach to the landing site.

To expand on the echo processor technology, a second phase of the research program was undertaken with the objective of developing and demonstrating the feasibility of a weather radar-based precision approach concept. The feasibility criteria for this concept included (1) minimal, passive, or battery-powered ground-based equipment; (2) the same pilot technique for flying the approaches as for instrument landing system (ILS) approaches; and (3) airborne weather/mapping radar modifications that could be accomplished as inexpensive retrofits for current civil radar systems.

To meet these objectives, a concept was pursued whereby an array of specially designed directional passive reflectors oriented along the localizer track would provide the directional signals necessary to derive ILS-type guidance. By using an on-board digital microprocessor installed in conjunction with the airborne weather radar, glide slope and localizer guidance would be calculated and displayed to the pilot on the existing ILS course deviation indicator (CDI). The reflector-based ground station would need no ground power, but would require 1.2 to 1.8 km (4000 to 6000 ft) of terrain for installation of the reflector array when used in conjunction with civil weather/mapping radar systems. Although this requirement would not be a problem for aircraft landing on conventional runways, it would be impractical for heliports.

An alternative to the radar reflector array, a radar transponder-based ground unit, has proven to be much more practical. An X-band transponder beacon with multiple-pulse reply capability was modified to reply through an array of directional antennas. This beacon-based ground station can be packaged in a small, inexpensive, battery-powered, portable unit.

In early testing of the BLS, the concept feasibility was demonstrated (Ref. 2). Current work is in progress with the objective of refining the BLS concept by (1) improving ground station design to eliminate multipath, (2) reducing localizer sensitivity at close ranges, and (3) preserving full-weather/mapping radar capability while on a BLS approach.

This BLS concept has significant potential for a large number of applications. It differs from other portable landing system concepts in that the airborne radar is actively used to interrogate and receive the ground station signals. Thus, distance to the landing site is inherently available on-board the aircraft. Also, the ground station power requirements are small because of the pulse-type replies of the ground station instead of the continuous wave (CW) mode of operation used in other landing systems. This paper describes the BLS concept, the concept demonstration hardware, and the flight tests in progress to verify the design principles.

CONCEPT DESCRIPTION

The BLS concept represents a combination of advanced, digital signal-processing techniques and X-band radar systems. Many of the same operating principles are used for a standard ILS, with important differences being in the carrier frequency and beam-discrimination methods. The following sections describe the operating theory and the concept demonstration hardware built to validate the feasibility of a weather radar-based precision approach concept.

Landing System Concepts

The weather radar precision approach concept operates on the principle of four overlapping, narrow radar beams oriented left, right, above, and below the desired flightpath. The sketch in Fig. 1 depicts the two glide slope beams, oriented above and below the desired flightpath. With this beam orientation, as the aircraft deviates from the desired flightpath, one signal increases in amplitude and the other decreases. When all four signals are of equal intensity, the aircraft is on course. Glide-slope deviation from the desired course is proportional to the difference in received signal strength of the up-down beams, and localizer deviation is similarly derived using the left-right beams.

A survey shows that two basic types of precision approach systems are used: Fixed-beam systems and scanning-beam systems (Ref. 3). Fixed-beam systems, including ILS and BLS, provide a single approach corridor, whereas scanning-beam systems, such as the microwave landing system (MLS), have the added flexibility of pilot-selectable approach paths (Ref. 4). A summary of ILS, MLS, and BLS characteristics is shown in Table 1. Although ILS and BLS are both fixed-beam systems, there are important differences between the two. First, the carrier frequency for the ILS beams is two orders of magnitude lower than for the X-band BLS. Since antenna size to achieve a given beam width is inversely proportional to carrier frequency, the high frequency of the BLS makes it possible to use small antennas at the ground site. Second, the techniques for discriminating between the four beams are very different. For ILS, the ground signals are transmitted on a CW basis, and they are tone-modulated for purposes of discriminating between the beams. The BLS makes use of the multiple-reply capability of X-band ground transponder beacons, incorporating a high-speed switching circuit to transmit the time-sequenced replies through the four directional antennas. The on-board microprocessor installed in conjunction with the airborne weather/mapping radar can then discriminate between the four directional guidance beams based on the time sequencing of the pulses. Unlike other landing systems, the BLS is a transponder-based system, and range to the ground station is inherently available. Other landing systems require co-located distance measuring equipment (DME) or marker beacons to provide the pilot with range fixes.

Ground-Based System

The ground station consists of a modified X-band radar transponder beacon with multiple-reply capability. Normally, the first reply is used to identify position, and additional time-sequenced replies are used for identification. In a standard beacon, all replies are transmitted through an omnidirectional antenna. Power for the beacon is either 28 V DC or 50-60 Hz, 120 V AC. With the BLS concept, the beacon is modified to the extent that a logic circuit is added into the normal beacon receiver video and modulator lines. This logic circuit is used to control both the beacon transmissions and a single-pole, five-position, solid-state microwave switch connected to the beacon transmitter. The switch allows sequential transmission of beacon reply pulses from five different antennas for each interrogation. The logic operates as follows. In the absence of an interrogating signal from an airborne weather radar, switch position 1 is selected, connecting the beacon to an omnidirectional antenna. Upon interrogation, the first reply pulse is transmitted through this omnidirectional antenna, providing a beacon-type wide-coverage reply for general landing site identification. After the first reply pulse, the logic circuit sequentially switches five beacon identification pulses, four to the four directional antennas, respectively, and one back to the omnidirectional antenna. The purpose of the last omnidirectional pulse is to provide a pulse spacing between it and the first pulse that positively identifies the station. The net result of this process is the radiation of six pulses, the first from the omnidirectional antenna, followed by four directional pulses, followed by another omnidirectional pulse.

As seen in Fig. 2, the four directional antennas used for the early concept demonstration BLS ground station were standard 30-cm (12-in.) weather radar flat-plate antennas. These antennas were chosen for test purposes because of their low cost and availability, but testing revealed multipath problems associated with such small antennas. A study of antenna size versus system performance indicated that BLS antennas 60 to 120 cm (2 to 4 ft) in height would be best. Current tests are being conducted with two parabolic antennas, 23 cm (9 in.) wide and 90 cm (36 in.) high, replacing the four directional antennas. Each parabolic antenna contains two appropriately oriented feed horns so that one antenna provides localizer beams and the other provides the glide-slope beams. These directional antennas should allow for multipath-free BLS operations at glide slopes of 4° or greater.

For current testing, a change in the design is being considered in which two sector antennas replace the single omnidirectional antennas. The sector antennas are higher gain, resulting in either reduced ground station transmission power or greater BLS range. Also, proper localizer CDI indications can be derived at deviations up to ±35° from the course centerline using amplitude comparison of the sector antenna replies. The current ground station, incorporating the two parabolic antennas and the two sector antennas, is packaged on a compact, portable pallet as shown in Fig. 3.

In early BLS testing, the ground station always operated in a transponder mode. However, in current testing, a dual-mode transmission is being used to preserve full on-board weather radar capability on approach. The transmission modes include the transponder, or synchronous, mode for operation during interrogation by the airborne radar and an asynchronous mode in which the ground station transmits pulse trains 100 times per second. In the asynchronous mode, the aircraft receives localizer and glide-slope guidance, but no range information can be derived.

Airborne System

The weather/mapping radar used for the BLS demonstration flight tests is typical of radars installed for offshore operations. The radar is an X-band (9375 MHz) color radar, with an average pulse power of 8 kW, and a pulse repetition rate of 121 pulses/sec. The radar can be operated in a primary mode, beacon mode, or a combined radar and beacon mode. For early BLS testing, the normal 46-cm (18-in.) flat-plate antenna was replaced with a very small, nonscanned, wide-beam antenna. Current BLS flight testing configuration includes a separate X-band receiver to allow simultaneous BLS and weather/mapping radar operation.

The BLS processor analyzes the beacon video signal to calculate range, localizer deviation and glide-slope deviation. Localizer and glide slope deviations are displayed to the pilot on an ILS-type CDI, and range

information is available on a panel-mounted digital display. The BLS processor is designed to easily interface with the airborne weather/mapping radar as shown in the installation diagram (Fig. 4). The processor is microprocessor-based with A/D (analog/digital) and D/A converters. Two signals, the beacon-receiver video and the modulator trigger, are input to the BLS processor from the radar receiver/transmitter (R/T) unit, and an automatic gain control (AGC) voltage is returned to the R/T. In the current airborne configuration (Fig. 5), a separate X-band receiver is used for the BLS guidance signals, with the processor controlling its AGC. The weather radar modulator trigger is still input to the BLS processor for range determination when the BLS is in its synchronous (transponder) mode.

System Operation

This section describes the concept demonstration BLS equipment in operation. Figure 6 shows an oscilloscope trace of the ground station beacon return with the six BLS reply pulses spaced at 6-μsec intervals. The BLS microprocessor is programmed to search this radar return for the two omnidirectional radar pulses 30 μsec apart. When consistent omnidirectional returns are received, the first is tracked and range gates are opened at each directional pulse location to measure signal strength. The first omnidirectional pulse is also used to adjust the AGC voltage, keeping the X-band receiver in its linear range and ensuring that side lobes of the directional precision guidance antennas do not generate false courses. For each guidance signal pair, the signal amplitudes are differenced, scaled, and filtered for output to the CDI.

With the current BLS dual-mode equipment, the ground station automatically switches between the synchronous and asynchronous modes. The airborne radar, sweeping a ±30° sector at a sweep rate of 24°/sec, interrogates the BLS ground station for about 0.3 sec of each 2.5 sec. Between these periods of synchronous operation, the BLS reverts to the asynchronous mode of operation to provide continuous localizer and glide-slope course deviation information.

FLIGHT TEST PROGRAM

The early concept feasibility BLS flight testing was conducted with the BLS ground station configured with four 30-cm 12-in. flat-plate antennas. These tests measured navigational and pilot tracking performance, and obtained qualitative pilot opinions of system performance. Current BLS flight testing, using the new ground station described above, has the following objectives: (1) identify any multipath problems, (2) obtain pilot comments on "course softening" at short ranges, and (3) qualitatively measure performance of simultaneous BLS and weather/radar operation.

Aircraft

The test aircraft is an IFR-equipped Sikorsky SH-3G helicopter (Fig. 7), the military equivalent of the S-61N. The SH-3G is a twin-turbine, five-bladed, single-rotor helicopter with emergency amphibious capabilities. The aircraft has a flying boat hull and two outrigger sponsons, into which the main landing wheels can retract. The rotor diameter is 19 m (62 ft), the gross weight is 8660 kg (19,000 lb), and the maximum airspeed is 120 knots. During flight testing, two pilots, the aircraft crew chief, and one to three experimenters were aboard. Experimental equipment and data acquisition system equipment were mounted on a rack in the cargo area.

Test Locations

The SH-3G helicopter is based at the NASA Ames Research Center, Moffett Field, California. System checkout and initial evaluation flights are made at Moffett Field, and quantitative data collection flights are made at the Crows Landing NALF, Patterson, California. The NASA Ames Flight Systems Research Facility at Crows Landing, equipped with radar tracking systems, a data telemetry receiver, and ground-based data monitoring and recording equipment, is used to record quantitative data to analyze BLS performance.

Approach Procedures.

The approach procedures being tested are similar to those used for standard ILS approaches. Tactical air navigation (TACAN) bearing and distance are used to position the aircraft for BLS intercept. Following acquisition of the BLS guidance, the warning flags on the BLS CDI disappear, and the pilot intercepts and tracks inbound on the BLS course. On system checkout and demonstration flights at Moffett Field, the BLS ground station is located near the approach end of runway 32R, and approaches are made parallel to the runway 32 approach corridor. Glide slopes ranging from 4° to 9° have been demonstrated. At Crows Landing, most approaches are made with the BLS ground station located 61 m (200 ft) left of the runway 35 centerline and near the STOL runway threshold. This location allows excellent tracking system coverage throughout the approaches.

FLIGHT TEST RESULTS

Figure 8 shows a typical view of the helicopter during early testing as it approaches the battery-powered BLS ground station on an approach. Note that during this early portion of the flight test program, the omnidirectional ground station antenna was replaced with a directional antenna to match signal strengths at the airborne receiver. For current testing, the directional transmissions are attenuated 10 dB to match signal strengths from the sector antennas. Testing to date has demonstrated BLS guidance intercept at ranges out to 17 n. mi. and glide slopes ranging from 4° to 9°. For quantitative evaluation, a 6.6° glide slope was used with localizer intercept 8 to 10 km (5 to 6 mi.) out from the ground station.

Pilot Comments

Pilot comments on the BLS have been favorable and enthusiastic, confirming the operational feasibility of the BLS concept. Pilot workload and piloting techniques are like those of ILS approaches. Since localizer and

glide-slope intercepts and course tracking use standard ILS techniques, pilot acceptability of the BLS approaches has been excellent, and pilot training on BLS approach procedures was minimal.

Pilot comments during early testing also identified the need to reduce close-range localizer sensitivity. A fixed-base piloted simulation was conducted to investigate pilot acceptability of several localizer "course softening" algorithms. These algorithms reduce localizer sensitivity as a function of range, which is available aboard the aircraft when the BLS is in the synchronous (transponder) mode. Results of the simulation identified a number of algorithms that reduced pilot workload at close ranges. Two of the most promising of these algorithms are being incorporated in the current flight tests for further pilot evaluation.

Pilot Tracking Performance

During three data flights on which the pilots were flying under simulated instrument meteorological conditions (IMC), 25 approaches were made. Composite pilots showing the lateral and altitude tracks are shown in Figs. 9 and 10, respectively. Figure 11 shows the one-sigma standard deviations of localizer cross-track errors achieved during this BLS testing. Also shown are the comparable envelopes for 6° glide-slope MLS approaches (Ref. 5, see Fig. 14) and airborne radar approaches (ARA) to oil rigs both with and without automatic target tracking equipment (Ref. 6). The one-sigma standard deviations from glide slope for BLS and MLS approaches are shown in Fig. 12. (Note that since ARA are nonprecision approaches, there is no glide-slope tracking data for comparison.) These envelopes show that the tracking performance achieved with the concept demonstration BLS was excellent, far exceeding that previously achieved for civil ARA and comparable to that achieved on MLS approaches.

BLS Navigation Accuracy

During early BLS testing, the BLS navigation accuracy was recorded. Although this equipment was in no sense optimized for accuracy, studies of the system errors are proving useful for further development of the BLS concept. Navigation errors identified to date include bias errors and signal multipath effects.

Bias errors, particularly in the localizer course, resulted from two sources: alignment of the directional antennas on the BLS ground station with respect to each other and alignment of the BLS ground station with respect to the desired approach course. For glide slope, use of an inclinometer for setting the ground station pallet at the desired glide slope was very repeatable, with repeatability achieved over the five data flights within ±0.1°. However, ground station glide-slope antenna alignment was 0.6° above the reference plane of the ground station pallet. Setup of the localizer course by siting along the edge of the ground station pallet was less accurate, and localizer course biases of up to ±2° occurred. For the current system, an improved localizer alignment siting method is being incorporated into the ground station.

Another problem, identified early in the test program, was with multipath, particularly in the glide-slope signals. The glide slope exhibited some waviness, which was worse at the lower glide slopes. Subsequent ground tests confirmed some nonlinearities in the approach course attributable to multipath phenomena. Reduction of the multipath errors was achieved using the current 90-cm (36-in.) high ground station antennas in place of the 30-cm (1 ft) antennas of the concept demonstation BLS.

Independent of the bias BLS error, the one-sigma navigation accuracy achieved with the concept demonstration BLS was ±0.22° in localizer and ±0.14° in glide slope. Figures 13 and 14 show composite data from the flight tests, comparing the localizer and glide-slope positions calculated by the BLS with the actual localizer and glide-slope deviations as determined using the tracking radar. These navigation accuracy data points were taken over a period of 3 weeks on four separate data flights.

CONCLUSIONS

A novel X-band precision approach concept has been successfully developed, and the concept feasibility has been demonstrated in flight testing. This concept appears to have significant potential for both civil rotorcraft operations and certain military missions in which remote-site precision landing systems are required. The portability and low power consumption of the BLS ground station also make the concept attractive for emergency and rapid deployment missions that require precision approach capability. Specific project conclusions are as follows:

1. The BLS X-band ground station is portable, compact, inexpensive, lightweight, and battery powered.

2. ILS-type guidance can be derived using a small microprocessor, easily interfaced with airborne weather/mapping radar or with a separate X-band receiver.

3. Pilot workoad and piloting technique for BLS approaches are similar to those in conventional ILS approaches.

4. Approach cross track errors using the BLS are far smaller than those achieved previously for civil ARA and comparable to those achieved on MLS approaches. Glide-slope tracking errors using the BLS are also comparable with MLS.

5. One-sigma navigation accuracy achieved with early concept demonstration BLS equipment was ±0.22° in localizer and ±0.14° in glide slope with bias errors of less than ±2.0° for localizer and ±0.1° for glide slope.

6. Simulation results showed that pilot workload at close ranges can be reduced by using an on-board algorithm for localizer "course softening."

7. Use of a 90-cm (36-in.) high parabolic antenna should allow for multipath free operation at glide slopes of 4° or greater.

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TABLE 1. COMPARISON OF LANDING SYSTEMS

System characteristics	ILS	MLS	BLS
Frequency	100 MHz (localizer 300 MHz (glide slope)	5000 MHz	9400 MHz
Antenna size	Large	1.8 to 3.6 m (6 to 12 ft)	0.6 to 1.2 m (2 to 4 ft)
Signal characteristics	CW, tone-modulated	Interrupted, CW	Transponder using sequential pulses
Guidance beams	Fixed: up, down left, right	Scanning	Fixed: up, down left, right
Derivation of guidance	Beam amplitude comparison	Time between signal peaks	Beam amplitude comparison
Range data	Requires co-located DME or marker beacons	Requires co-located DME or marker beacons	Inherently available
Airborne equipment	Widely installed	Must be added	Minimum retrofit for radar-equipped aircraft

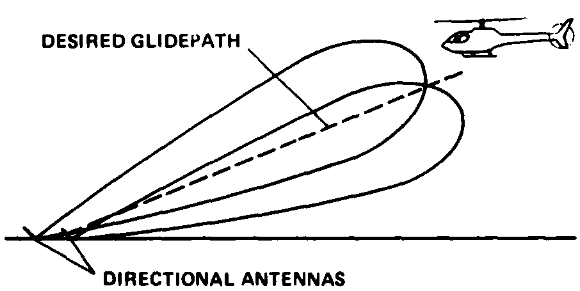

Fig. 1 Overlapping directional antenna beams provide course guidance.

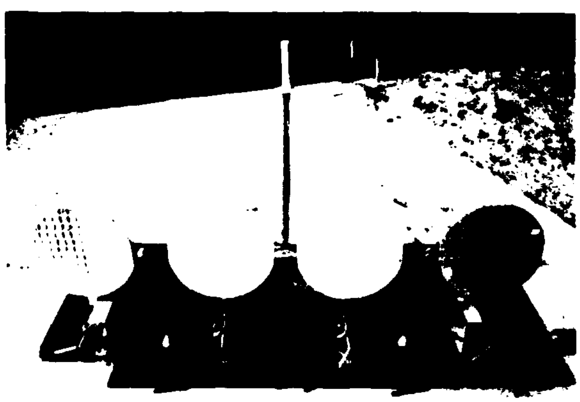

Fig. 2 Landing system ground station.

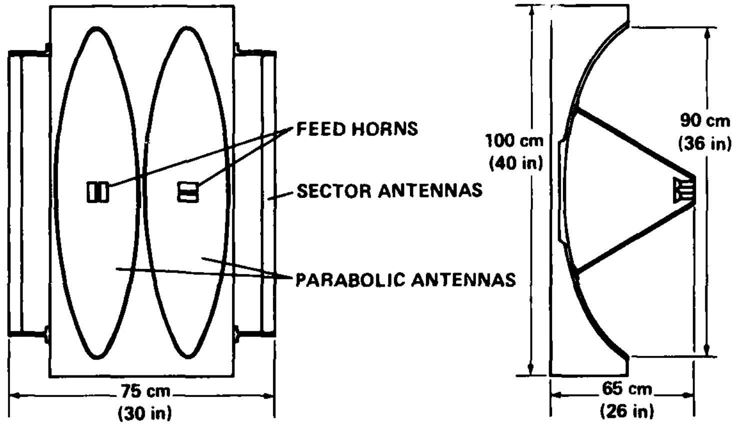

Fig. 3 Current parabolic antenna.

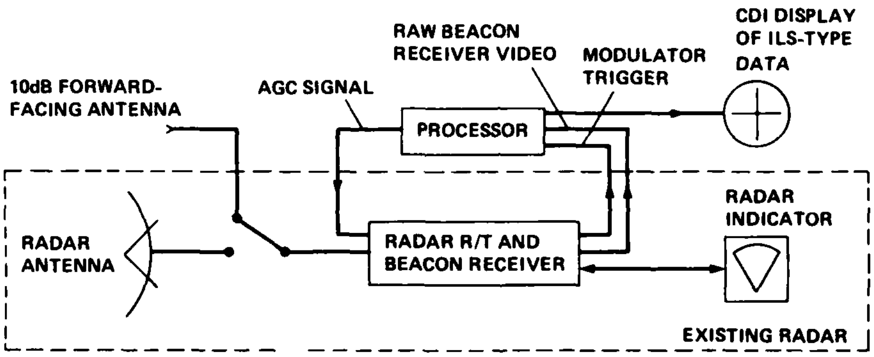

Fig. 4 Initial airborne configuration.

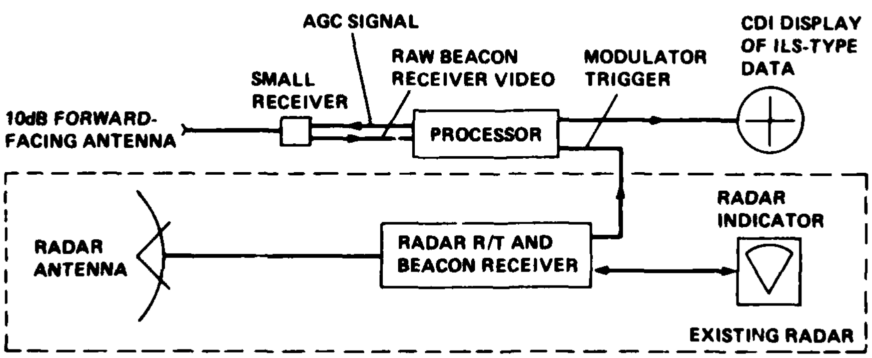

Fig. 5 Current airborne configuration.

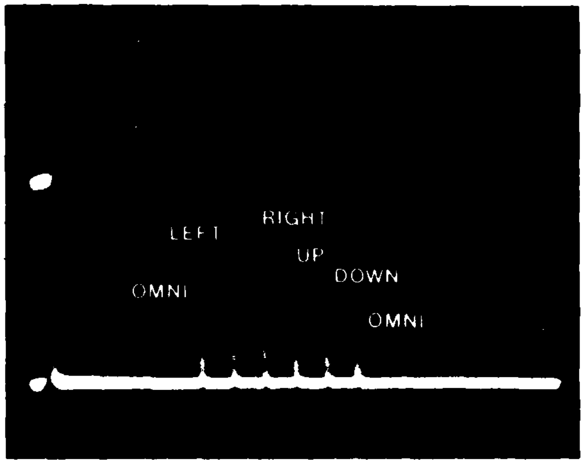

Fig. 6 Received beacon video signal aboard the test aircraft.

Fig. 7 Test aircraft.

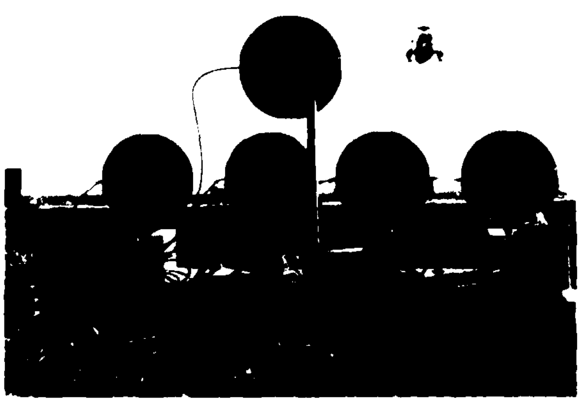

Fig. 8 BLS flight demonstration on short final approach path.

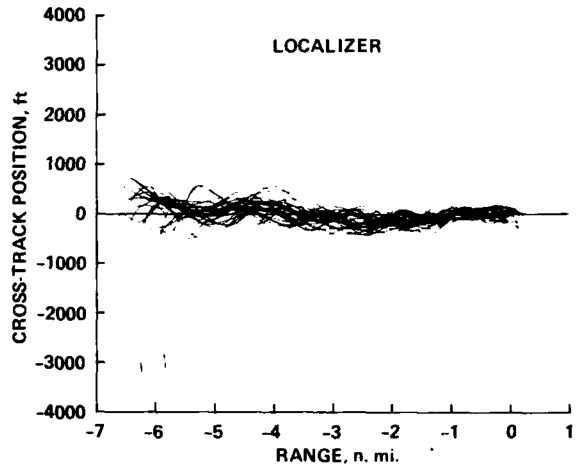

Fig. 9 Composite of x-y tracks on BLS approaches.

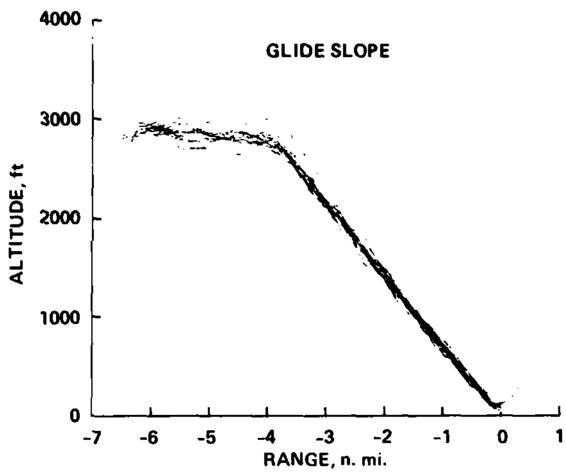

Fig. 10 Composite of altitude profiles on BLS approaches.

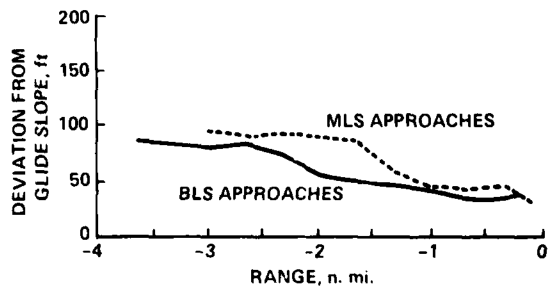

Fig. 11 Standard deviation of cross-track errors for various types of approaches.

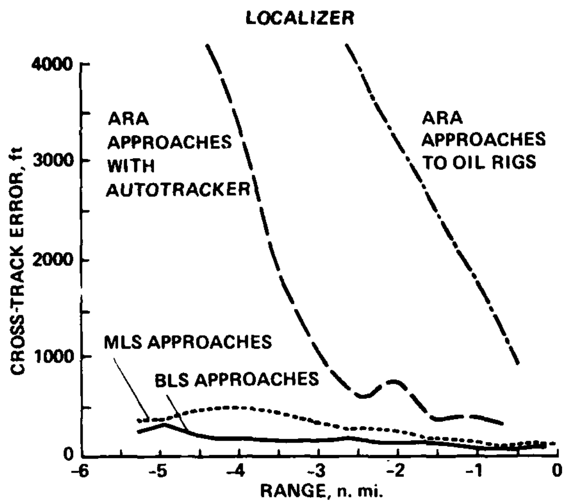

Fig. 12 Standard deviation of glide-slope errors for BLS and MLS approaches.

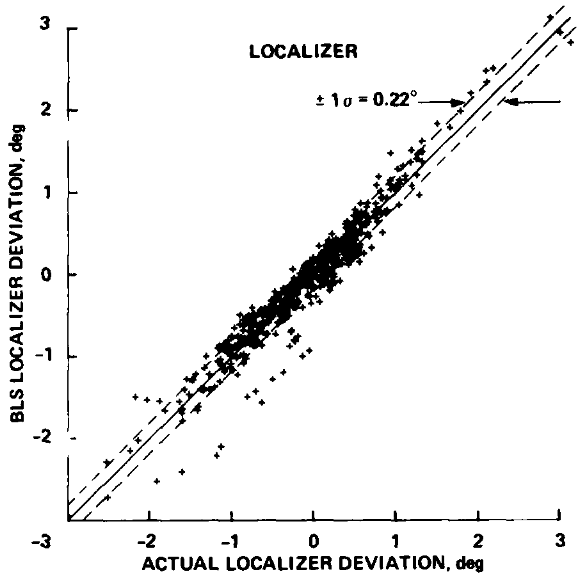

Fig. 13 Composite showing BLS localizer navigation accuracy.

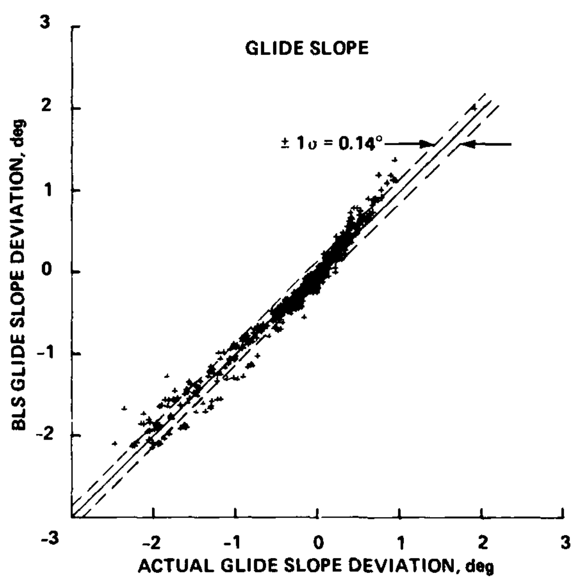

Fig. 14 Composite showing BLS glide-slope navigation accuracy.

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14. Abstract

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- Requirements and operational use of future helicopters including reconnaissance, weapon delivery and logistic support.
- New methods of briefing helicopter crews and tactically setting up the mission systems before and during the flight.
- Flight control, displays and tactical performance management, navigation.
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